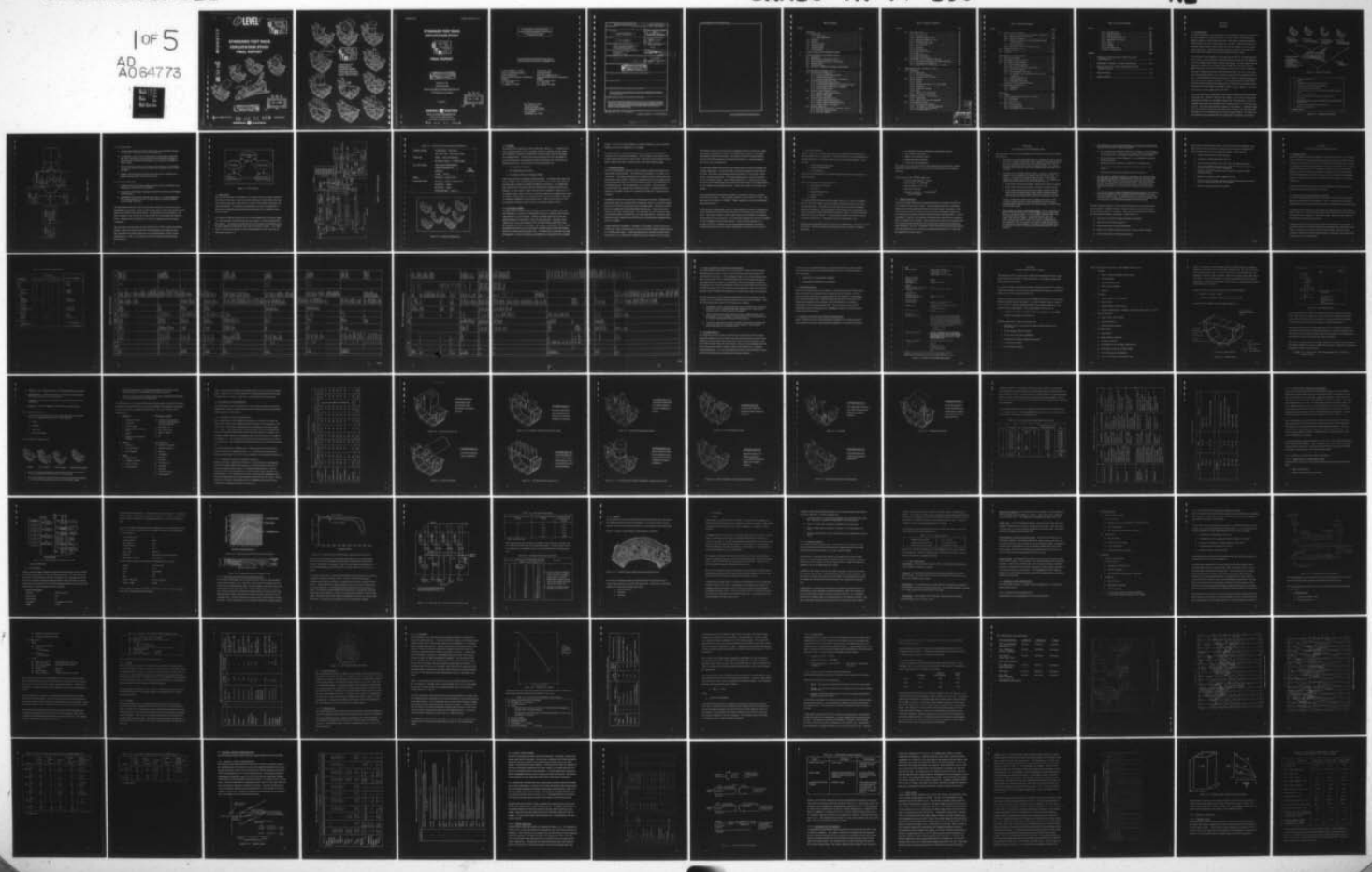
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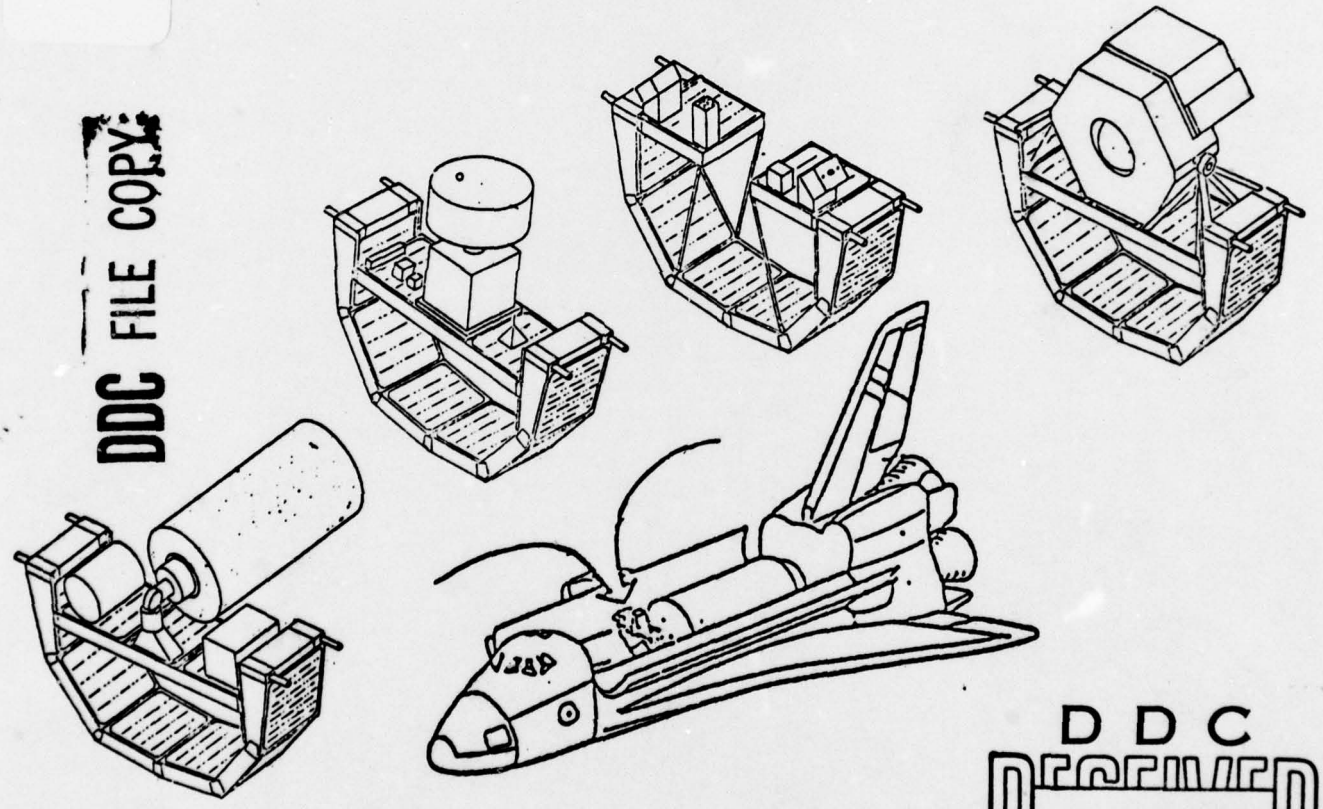
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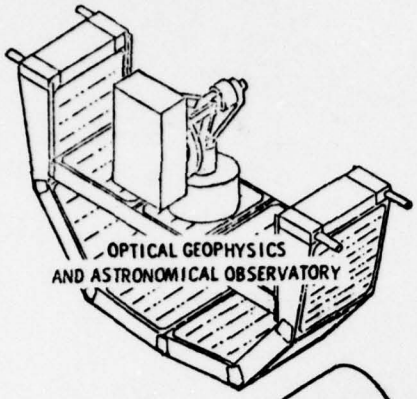
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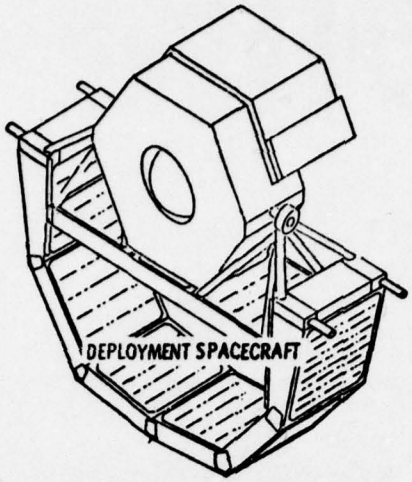
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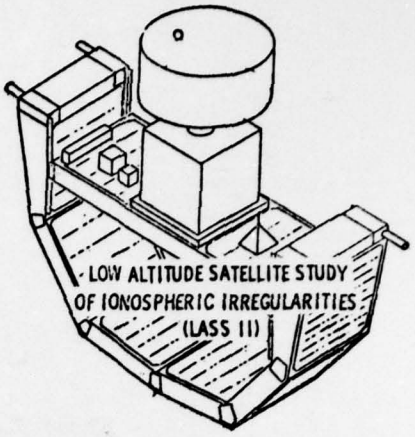
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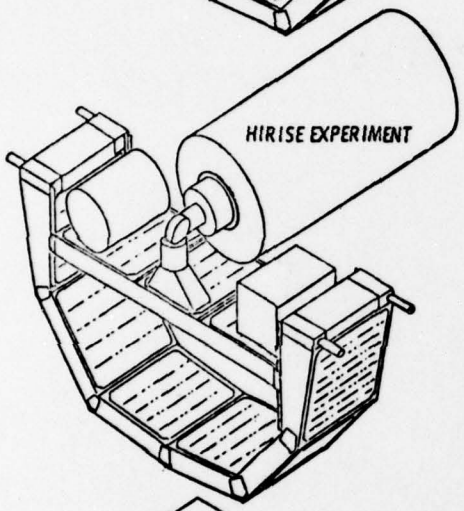
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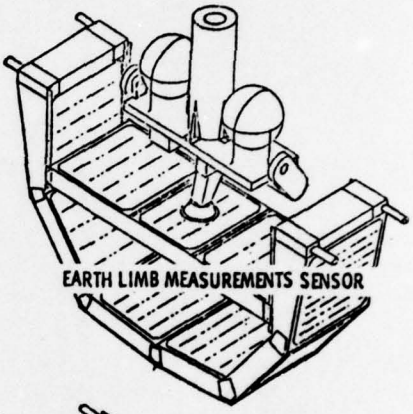


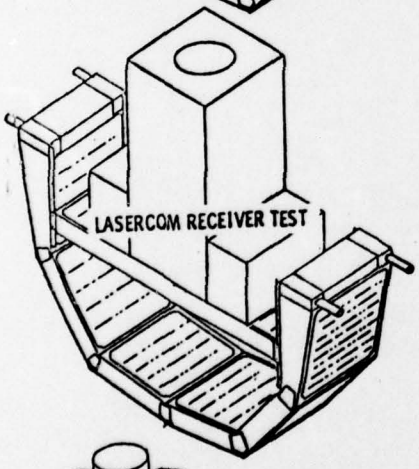


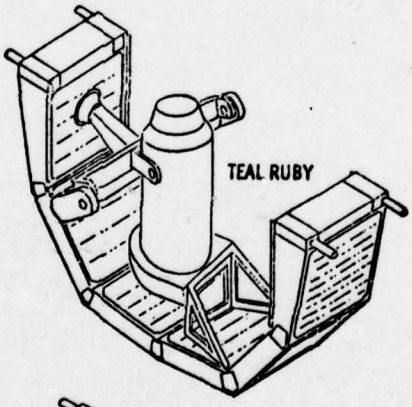


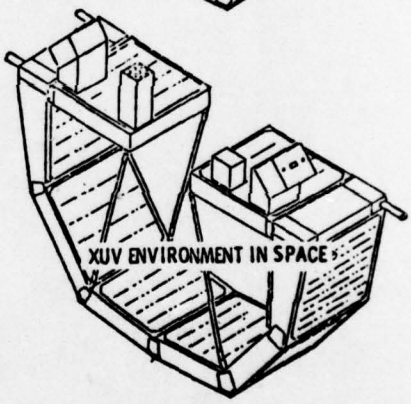


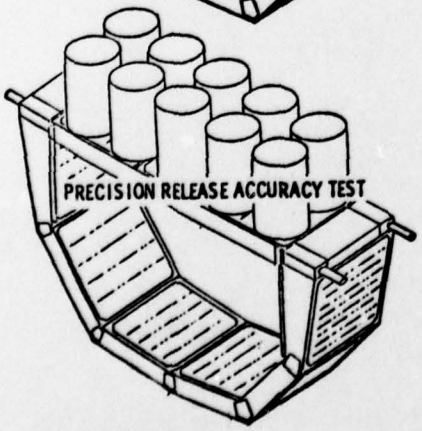
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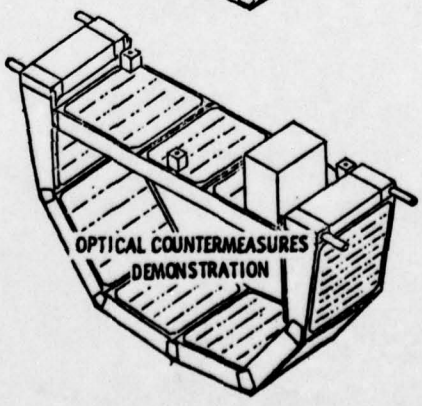


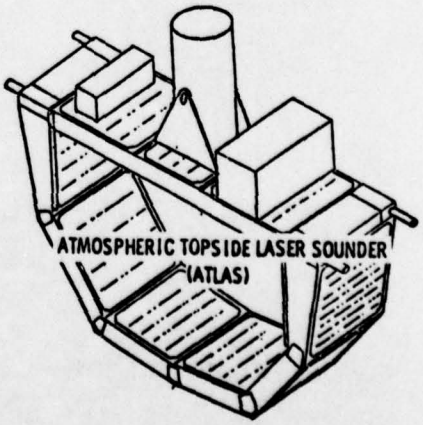












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The Standard Test Rack (STR), Designed To Handle A Wide Variety Of Experiments and Operational Payloads, Can Fly On The Space Shuttle With A Minimum Of Lead Time and Within A Small Space Envelope. It Is Configured To Use Left-Over Space Including Fitting Around The Space Lab Tunnel. Minimum Interface With The Shuttle Support Systems Provides Ability For Quick Reaction Installation on A Space Availability Basis.

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SECTION 1
SUMMARY

1.1 INTRODUCTION

The primary design and operations goal for the Space Shuttle Program is to provide low-cost transportation to and from earth orbit. Spacelabs will be carried aloft by the Shuttle in support of manned orbital operations. Free-flying or automated satellites will be deployed and recovered from many types of orbits. Automated satellites with propulsive stages attached will be deployed from the Space Shuttle and placed in high-energy trajectories. Experiments and operational test missions will be flown in the sortie mode where testing can be performed without leaving the shuttle.

This approach to space operations will provide many avenues for conducting investigations in space. Many participants, representing diverse backgrounds and capabilities will work routinely in these space operations of the future. The Space Shuttle Era will emphasize operational reuse of flight hardware, which will result in low cost per flight to the users. Low cost was and continues to be the basic concept on which the total Space Transportation System (STS) is being developed. One way to achieve minimum costs during the Shuttle Era is to provide a relatively small, standard system to act as a buffer between the STS and small payloads (3000 lbs). Detail acquisition and Life Cycle Costs are provided in Section 7. This system maintains minimum Shuttle interfaces while providing housekeeping services to experiments/operational payloads. The Standard Test Rack (STR) is that Shuttle "buffer" system. Figure 1-1 shows the utilization and various configurations of the STR.

This study addressed the applicability of the Standard Test Rack for DOD type experiments/payloads. Candidate experiments (See Figure 1-2 were used to synthesize the requirements baseline for a preliminary design of the STR subsystems. The twelve candidate experiments were identified in the Aerospace letter E.P. 77-027 dated June 21, 1977. The study approach shown in Figure 1-3 was followed. A structural design and STR integration and operation procedures were developed. The STR system and life cycle costs were estimated based on assumed mission profiles. All of these were accomplished while maintaining the study design goals and objectives shown below.

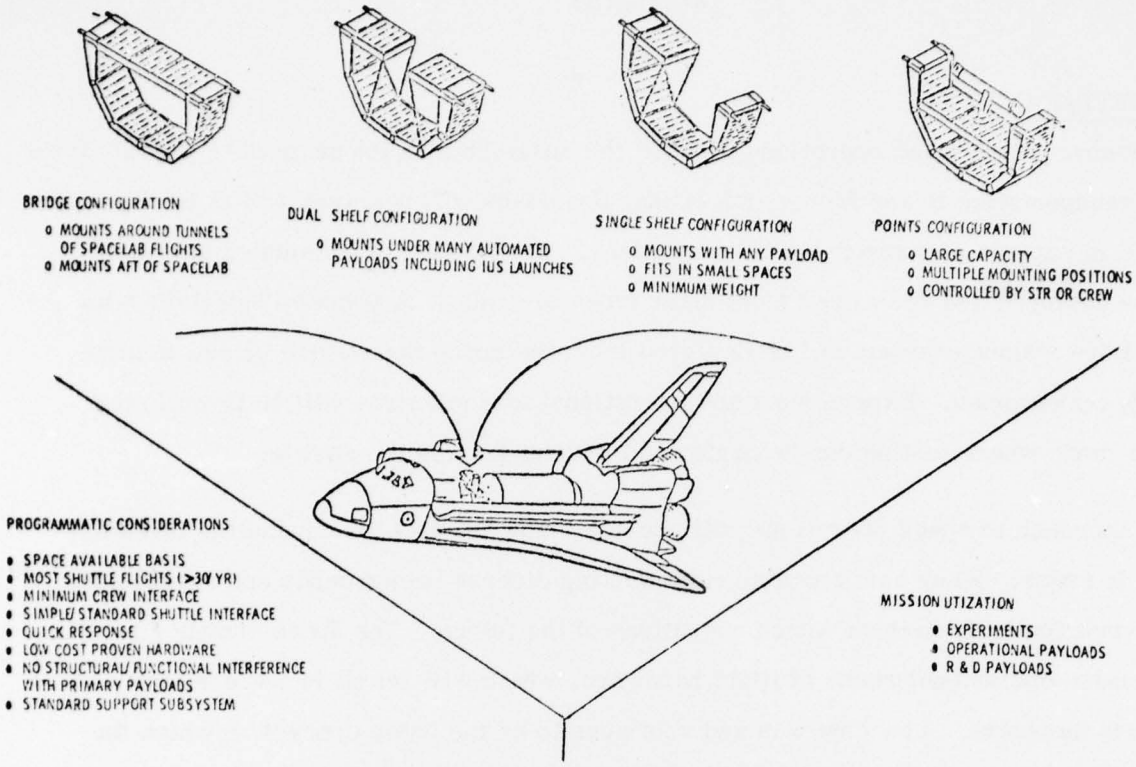


Figure 1-1. Standard Test Rack

- LASERCOM RECEIVER TEST
- HIRISE*
- ATMOSPHERIC TOPSIDE LASER SOUNDER (ATLAS)
- PRECISION RELEASE ACCURACY TEST (PRAT)*
- EARTH LIMB MEASUREMENTS SENSOR*
- LOW ALTITUDE SATELLITE STUDY OF IONOSPHERIC IRREGULARITIES (LASS II)
- XUV ENVIRONMENT IN SPACE
- OPTICAL GEOPHYSICS AND ASTRONOMICAL OBSERVATORY*
- TEAL RUBY*
- PHYSICS INTERNATIONAL MODULAR BREMSTRAHLONG SOURCE (PIMBS II) EXPERIMENTS
- OPTICAL COUNTER MEASURES DEMONSTRATION
- DEPLOYABLE SPACECRAFT

*REQUIRE GIMBAL

Figure 1-2. Candidate Experiments

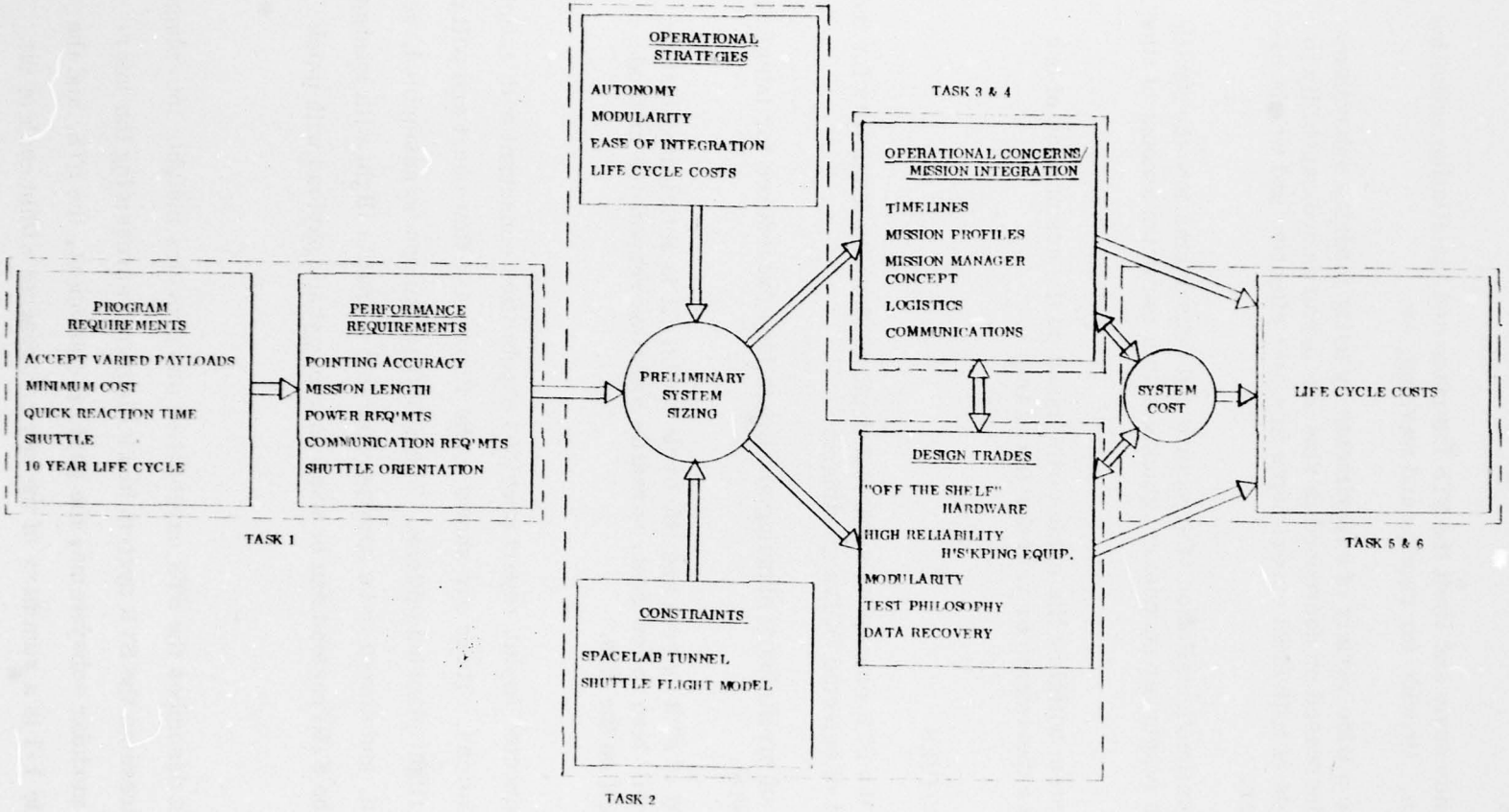


Figure 1-3. Study Approach

1.1.1 STUDY GOALS

1. Operate autonomously and limit the STS interface and installation activities to a minimum. Provide for caution and warning, etc.

2. Accommodate a wide variety of experiments by using modular subsystems that may be increased or decreased in size and performance capability to meet the needs of individual experiments in a cost effective and weight efficient manner.

3. Permit integration/installation of structure and subsystems into the Shuttle bay and cabin (whenever necessary) quickly with a minimum amount of time and effort.

4. Develop a system with low life cycle costs based on the assumption of one flight of each experiment per year for ten years.

1.1.2 STUDY OBJECTIVES

1. Definition of the STR concept in sufficient depth to show its feasibility for the flight support of selected DOD experiments.

2. Development of preliminary planning and operations procedures for integration of STR with STS

3. Establishment of STR system and life cycle costs. It is anticipated that the STR concept will lead to low cost, versatile and time responsive methods for space flight on the STS.

The STR design as delineated in this report provides capabilities consonant with anticipated Shuttle Era operations. These are shown in the Figure 1-4 flow chart and will provide planners with flexible flight planning options. The STR/payload can be assembled, verified, acceptance tested, and stored to be incorporated into a Shuttle flight with minimum notice. Additionally the STR/payload can be used as a piggyback payload with quick reaction time.

The body of this report describes the STR capabilities and provides insights into planning options. Figure 1-5 presents the STR system block diagram representing the interfaces with each of the modular subsystems, the payload equipment, the STS, and the ground systems. Table 1-1 is a summary of the key components within each of the STR subsystems.

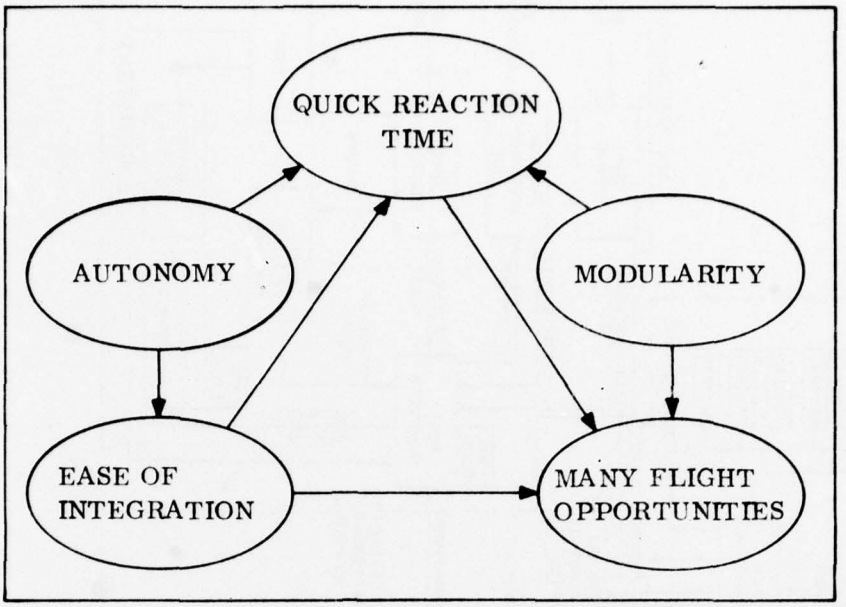


Figure 1-4. STR Features

1.2 STRUCTURE

The structure subsystem was designed to use standard repetitive members, high allowable piece-part tolerances, common materials (aluminum), formed sections and minimum tooling requirements, high margins of safety to provide infinite fatigue life, and replaceable fittings. All component mounting panels were made interchangeable to allow for use of multiple battery and thermal control panels and to maintain a center of gravity control.

The structural design also permits the use of varied configurations as shown in Figure 1-6, which maximizes the utility of the STR by providing a versatile experiment interface/envelope. The design is such that the STR can be installed to fit around the spacelab tunnel utilizing available Shuttle space that was heretofore unusable. As an added feature, the STR can be installed after the Sapcelab tunnel is in place in the STS as described in section 4.5.2.

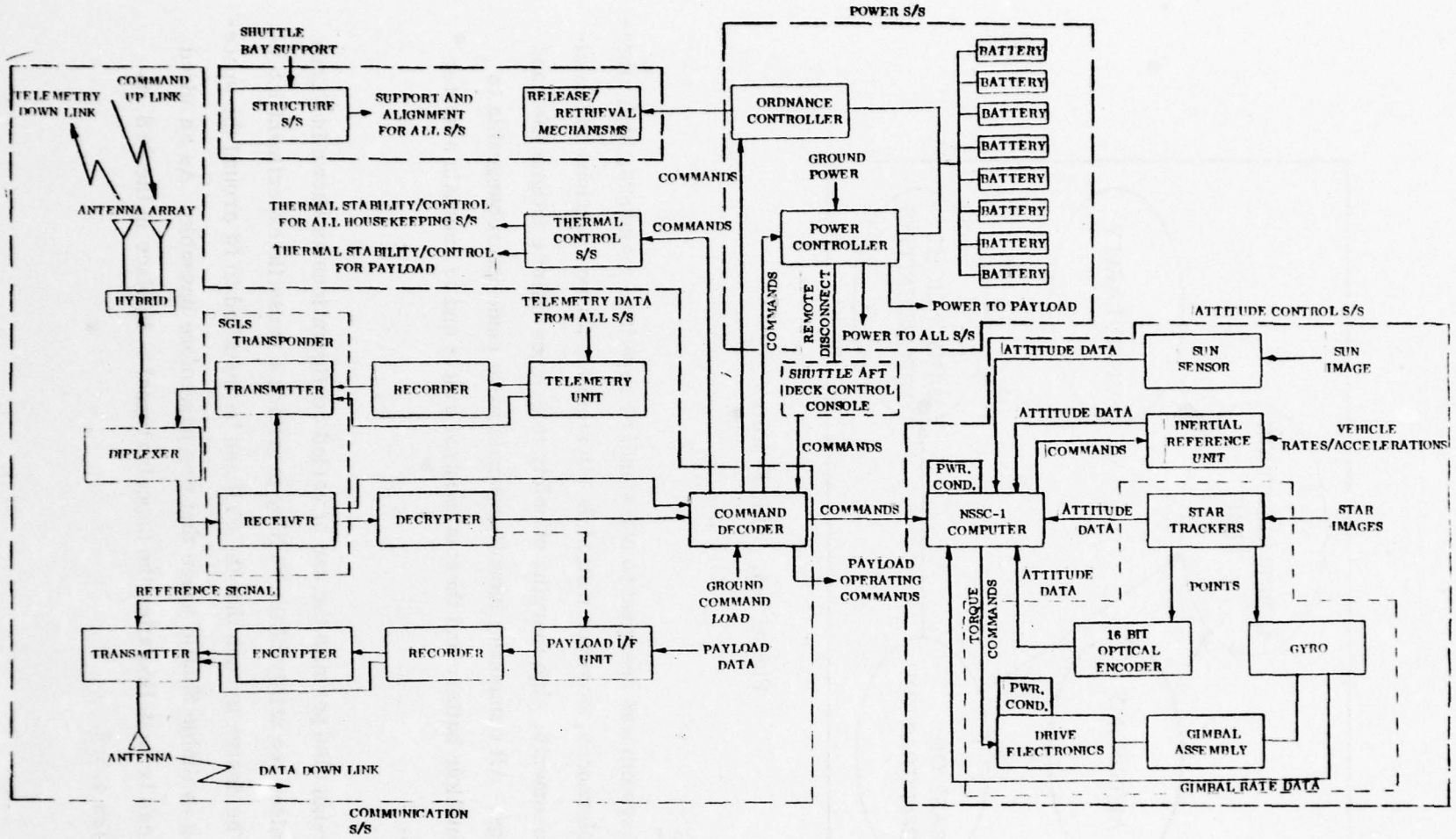


Figure 1-5. System Block Diagram

TABLE 1-1. Summary of Recommended STR Subsystems

THERMAL CONTROL	DISTRIBUTION - HEAT PIPES HEAT REJECTION - MULTIPLE OPTIONS	
STRUCTURE	FRAME - BUILT UP ALUMINUM EQUIPMENT PANELS - STIFFENED SHEET	
ATTITUDE CONTROL	FIXED HEAD STAR TRACKERS INERTIAL REFERENCE UNIT COMPUTER GIMBAL - POINTS	NASA STANDARD
POWER	BATTERIES - $LiSO_2$ PRIMARY	
COMMUNICATIONS	RECORDER - NIMBUS G/ SEASAT/ MMS ENCRYPTOR - KG28A DECRYPTOR - KG23A OTHER EQUIPMENT - DSCS III	

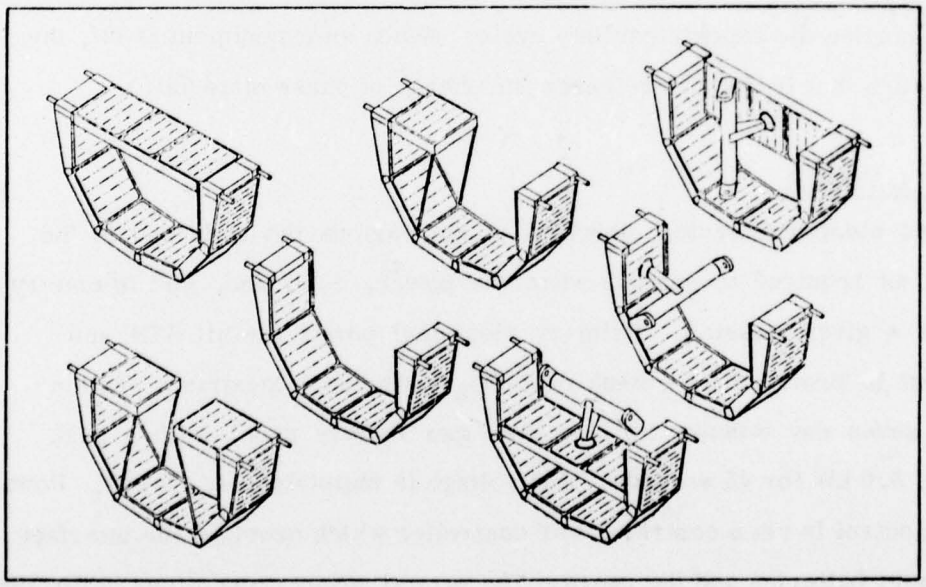


Figure 1-6. Structural Configurations

1.3 THERMAL

The thermal control subsystem consists of heat pipes which are incorporated as part of the component panels and main strongback channels to distribute the heat, multi-layer insulation to maintain an "isothermal" cocoon, and three separate approaches for rejecting the heat. The three heat rejection schemes have been investigated in some detail and all are recommended approaches depending on the experiment to be integrated on the STR. These three heat rejection approaches are:

1. Use shuttle heat exchanger
2. Use variable heat pipe radiator
3. Use change of phase heat exchanger module

Each of the above has its advantages and disadvantages. The Shuttle coolant loop is not always available at the required temperature because of use by other prime payloads; but when usable, provides essentially an isothermal environment. The variable heat pipe radiator must be integrated with other payloads and experiments (field-of-view blockage, etc.) but provides a large total heat rejection capability (~.26kw per 20 ft.^{2}). The change of phase module provides an essentially isothermal environment and is easiest to integrate, but has a finite heat rejection capability (~15 kwh per module). This could be extended by changing the experiment duty cycle. (When an experiment is off, the heat transfer is such that it tends to refreeze the change of phase material.)

1.4 ELECTRICAL POWER

The STR provides electrical system services to the various payloads and to the STR subsystems as required to accommodate the power, command, and telemetry requirements for a given mission. Primary electrical power for all STR and payload equipment is provided by a bank of $LiSO_2$ batteries. Maximum system capability for a seven day mission is 37.8 kWh (per battery panel) with a peak load capability of 3.0 kW for 45 seconds. Bus voltage is maintained at +28±4V. Power distribution and control is via a central power controller which provides the interface between the primary batteries and the users. The ground power connections is through this equipment. All power switching is accomplished by commands from the command

decoder. STS crew override capability is provided to disable any power controller functions in the event of malfunction or emergency.

All payload equipment requiring other than 28±4 volts power must provide self-contained power converters and regulators. These secondary power systems will be electrically isolated from the primary bus. Both primary and secondary systems will be isolated from vehicle structure except at the vehicle unipoint ground.

1.5 COMMUNICATIONS

The STR communication subsystem provides command storage and decoding for all commandable system functions. All commands are executable in "real time" or in "stored time". Real time commands may be executed by ground control at any time during ground station contact. In addition a command message can be loaded into the on-board command decoder memory for execution at a later time as indicated by accompanying time tag. Time tag granularity is 0.2 seconds. On-board storage can accommodate 256 commands with attendant time tags. Command formats and operation is compatible with the 20 bit capability of the KIR-23A command security equipment.

Capability for 600 discrete commands in matrix format is provided. Individual loads and typically relays are energized by the simultaneous selection of one of 30 X drivers and one of 20 Y drivers as determined by the command format. All loads are located in the user equipment but electrically isolated from the user. They are to be considered as part of the command system. User interfaces are via the contacts of the selected relays which are considered as part of the using system. Isolation is maintained between the command system and all users to prevent failure propagation from one user to another.

Variable digital data commands or "messages" are provided to all users via a common data bus. These commands may be used for any function capable of being expressed as a variable binary number. Addressing information is contained within the serial message and up to 14 variable bits of information can be provided to any of 16 users.

The telemetry portion of the communication subsystem provides the buffering, signal conditioning, formatting and storage for all downlink data necessary for the proper functioning of the STR and its payload. These data are typically housekeeping data related to the health and status of the supporting subsystems. Operational payload data, if any, are voted separately.

For maximum flexibility the telemetry has variable format and data rate capability on a flight-to-flight basis. On any given flight for any given payload the format and vote are fixed. The format accommodates analog data encoded to 8 bit precision, bilevel discrete, and serial digital data. The typical downlink format comprises 30 frames at 256 8-bit words per frame. Data is PCM encoded NRZ-L at one kilobit per second. The frame rate is therefore one frame every 2.048 seconds with 61.44 seconds required for one complete cycle through the format. Sample rates as high at 16 per frame are possible.

An on-board telemetry/command tape recorder is privided for telemetry recording outside of station contact. Total recording capability is on the order of 10^8 bits. Playback rates can be as high as 1.024MBPS. Digital capability for real time and playback telemetry is provided.

A dedicated data handling system exists for those payloads that cannot be accommodated by the STR telemetry system because of data rates, data quantity or security requirements. This system provides a payload unique interface unit designed to accommodate the perculiarities of a given payload. An on-board data handling tape recorder is provided for data recording when out-of-station contact. Total recording capability is on the order of 10^9 bits. Capability for real time and playback telemetry is provided. A dedicated downlink service is provided containing data security devices if required.

1.6 ATTITUDE CONTROL

Analysis of the twelve candidate experiments in the mission model reveals that the Shuttle capability must be augmented. To meet the orientation, attitude knowledge and pointing requirements, an attitude control system is used which consists of 3 basic sections:

- o Attitude Determination System
- o Payload Pointing System
- o Control Software

Each section is moduliarized to be included during offline integration as required for payload operation.

The attitude determination system consists of:

- o Fixed Head Star Trackers
- o Sun Sensors
- o Inertial Reference Unit
- o Computer & I/O
- o Software

For payloads which need only attitude knowledge, this is all that need be provided by the STR. The Star Trackers are used to update the inertial reference unit to refine long term Gyro Bias'. The updated gyros maintain short term attitude knowledge between star encounters. The best estimate of STR attitude is then maintained in the computer, based on a mission dependent star catalog. However, during periods when bright objects come into the field of view of the star trackers, sun sensors are used to bound the attitude error. Knowledge/accuracy is approximately 1 min.

Payload motion independent of Shuttle maneuvers is controlled by a 2 axis gimbal system. This system, "Payload Orientation and Instrument Tracker for Shuttle" (POINTS), consists of:

- o An orthogonal structure suspended on precision ball bearings
- o Drive assembly electronics
- o Angular rate & position sensors
- o Command & control computer
- o Control and interface software

The computer is the same one used for attitude determination with modular software. Modularity provides flexibility for mission peculiar control software and specific interfaces.

Salient features of the "POINTS" system are:

- o Payload weight capability 1500 lbs
- o Rate accuracy 80 RAD/SEC
- o Gimbal freedom -60^{o} to $+90^{o}$
- o Slew rate 2 DEG/SEC
- o Nominal acceleration 4 1/2 DEG/SEC^{2} (1500 LB Payload)

1.7 CREW INTERFACES

Caution and warning indications and command capability are provided to the STS crew via the Shuttle aft deck control console. Selected telemetry monitors describing the health of the STR and payload will inform the crew of the status of the system. Sufficient information will be provided for the crew to make an independent judgement as to the operation and safety of the system without the benefit of ground contact. If direct intervention is required in the operation of the system, the crew will have two paths by which to implement their decisions. There will be a direct input from the console to the STR command decoder enabling the crew to execute any command in the command decoder repertory in real time. In addition a number of hardwired override type functions will exist enabling the crew to shut down any malfunctioning system or to circumvent any potentially hazardous situation.

SECTION 2

CONCLUSIONS AND RECOMMENDATIONS

The conclusions drawn from this investigation of the design and utility of the Standard Test Rack are:

1. The design and repeated operational usage of a Standard Test Rack is feasible and is a cost effective concept (This report contains a preliminary design and life cycle cost analysis of the STR and its related subsystems). This is achieved by application of state-of-the-art design practice, use of off-the-shelf flight proven hardware, high safety margin design, and minimization of testing.

2. The STR system and design provide autonomy, modularity, configuration flexibility, ease of integration into Shuttle, and low life-cycle costs. As a result, it is capable of accepting for flight on STS, a wide variety of experiments, prime test units and operational systems. The modular and configuration flexible STR can accept experiments that vary:

   a. From small hard mounted packages to large gimbal mounted equipment

   b. From sortie mission type systems to extended or deployed systems

   c. From simple exposure type missions to complex missions with high power and energy requirements, highly accurate pointing and slew requirements, tracking of points on the ground, high data rates with both on-board data storage and/or real time transmission to ground.

   All this is accomplished with maximum autonomy with respect to STS resulting in simple interfaces, shortened integration cycles and quick turnaround flight schedule potential, with the STR acting as a buffer between the STS and the payload.

3. STR is capable of "fitting" on most Shuttle flights. With its relatively compact size, the STR can fly on STS Spacelab flights, free-flier deployment flights, or combination flights. On a full up Spacelab Shuttle flight it can fit around the Spacelab tunnel, or even around the inter-pallet and module-pallet interface bridges. With the STR shelf configuration, it can fit around the spacecraft. When "open" space is available between other payloads it takes up a minimum of cargo bay length allowing maximum potential for payload changes. The result of the STR design concept is to maximize the number of potential STS flight opportunities for DOD equipment.

4. The STS/payload on-line integration timelines are minimally extended by STR (Approximately 10 hours). This is achieved by:

   a. Use of standard STS attachments (the STR is designed for quick attachment of equipment bridge or shelves to the strongback where piecemeal installation is required (i.e., installation around an inplace spacelab tunnel)).

   b. Electrical systems testing is designed to be accomplished by automated checkout equipment.

   c. Integration and test of experiments and STR is an off-line activity

   d. Ease of integration into the on-line activities such as CITE or SIVE testing

   e. Minimization of last minute servicing activities on the pad

5. No extraordinary equipment, facilities, or procedures are required to integrate the STR into the STS. A dedicated STR preparation, processing, and pre-flight storage area at WTR and ETR permit flight schedule flexibility and allows for "last minute" DOD equipment flights. With a factory-to-pad concept, this dedicated area can be located off-site if compelling reasons dictate this approach (i.e. lack of room for a dedicated area at the launch site).

6. Low life cycle costs are achieved using the STR system concept. The repetitive use of the functionally efficient, compact STR system tends to minimize the hardware and flight costs. The integration costs are minimized by application of the STR Mission Manager concept which utilizes centralized authority, dedicated contractor, and standardized procedures and documentation.

The above conclusions appear favorable towards implementation of a program to provide an operational Standard Test Rack. However, there are several areas or key issues requiring additional investigation. These deal with integration of the STR into the STS and affect the degree of autonomy. These issues are:

1. Assurance of STR/STS development schedule compatibility

2. Effect of RF in the Orbiter cargo bay

3. Latest possible date for flight committment

4. Caution and warning requirements and crew override of STR commands

5. Crew participation in STR/Payload operations

Based on the conclusions and key issues, a list of tasks have been identified. Completion of these tasks will provide resolution of the key issues and complete some of the necessary steps in the STR system development. These tasks are:

1. Complete detail structure design drawings
2. Fabricate an engineering model of the STR
3. Perform static and dynamic load tests on the STR engineering model
4. Carry out in-depth design and analysis of the STR subsystems emphasizing RF effects which must be Shuttle compatible
5. Perform a STR/STS integration cycle analysis for a specific mission including recycling
6. Integrate an experiment such as HIRISE on OFT-6
7. Perform mission and flight operations analyses to determine the interaction effects of STR payloads and other STS payloads
8. Prepare a preliminary STR users manual.

SECTION 3

STR REQUIREMENTS SYNTHESIS (TASK 1)

3.1 INTRODUCTION

The twelve candidate experiments identified in the Aerospace letter E.P. 77-027 dated June 21, 1977 and the subsequent information provided at the STR Concepts Definition meeting of July 15, 1977, were used as the baseline material for this study. However, it is recognized that these twelve experiments are only representative of many current and future experiments that may be adaptable to the STR concept. Based upon this knowledge, a modular concept of packaging STR equipment has been identified that permits interchangeability of panels within any of many STR structural combinations. This allows sizing the STR system to meet individual requirements for all baseline experiments.

The results of the subsystem studies and recommendations of selected concept approaches are presented in Section 4 of this report.

3.2 IDENTIFICATION OF EXPERIMENT DRIVERS

The twelve candidate experiments were reviewed and the STR system requirements and design concept drivers were identified. Table 3-1 presents the STR subsystem characteristic and the associated experiment which sizes that particular characteristic.

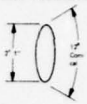
A summary of the STR subsystem requirements for each of the twelve experiments is given in Table 3-2. The most stringent values were identified for each STR subsystem and were utilized in sizing them. The details of these "envelope" requirements are discussed for each subsystem in Section 4.

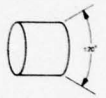
In general, all but one of the experiments appear to be capable of STR mounting. Experiment 10, Physics International Modular Bremstrahlung Source (PIMBS II) Experiment, is a special case because of its large size and weight. None of the other experiments present any currently identified special requirements that preclude their mounting on a STR for Shuttle flights.

Table 3-1. STR Design Concept Drivers

EXPERIMENT REQUIREMENT	STR SUBSYSTEM						DRIVING EXPERIMENT
	Struc.	Therm.	ACS	FOV	TT&C DH	Power	
Weight	X						HIRISE
Size	X						HIRISE
Power							
Continuous						X	ATLAS
Peak						X	PIMBS II
Total Energy						X	ATLAS
Thermal							
Supply	X	X					ATLAS
Distribution	X	X					TEAL RUBY
Dissipation	X	X					LASER COM
Rejection	X	X					
Communications							
Commands					X		TEAL RUBY
Data Transmission					X		TEAL RUBY
Recording					X		LASER COM
Attitude Control							
Knowledge			X				PRAT
Pointing	X		X	X			ELMS
Stability	X		X	X			TEAL RUBY
Special Items							
Field of View	X			X			XUV ENVIRONMENT
Payload Specialist Need			X	X	X	X	10 OF 12 EXPERIMENTS

Table 3-2. Experiment Requirements

No.	Title	Total Weight	Weight Break Down	Size	Duty Cycle	Astronaut/ Specialist Participation	Experiment Power	Thermal	Communications	ACS	FOV	Special
1	Lasercom Receiver Test	1050 lbs.	500 lbs. 500 lbs. 50 lbs.	Optics-4' x 4' x 8' Electronics-40^3 ft Control-1' x 1' x 2' (Loc. in S/C Cabin)	• 30 minutes each test for 2 tests per day on consecutive orbits • 7-14 days	Specialist to work 4-6 hrs. per day on or with experiment	• 28 ± 4 Volts • 1 kW peak during each test • 100 Watt stand-by • ~23.1 kWhr max.	Approx. 1 kW heat dissipation by radiator at 10-30°c. None during stand-by	• 64 kbps housekeeping VIA orbiter to ground during each test period • Up to 1 Mbps recorder DUMP VIA orbiter to ground twice a day (non test periods) • 1 kbps command VIA orbiter to ground prior to each test period and during tests	• ± 1.0 deg. for each 30 minute test along optical ref. line for alignment • Optical angular rate less than 0.01 deg/sec. all axes for stability • Optical angle and rate to 0.1 deg. and .006 deg/sec. respectively along optical ref. line for knowledge • Laserbeam (1.06) is beacon pointing and transmits tracking data to satellite at 10 kbps on the beam • Satellite transmits high rate (1 Gbps) from signal generator on a laser beam of 0.53 wave length recorded by the experiment at IMbps on tape recorder		• Crew not to look at incoming or outgoing laser beam with naked eye • Orbiter bay not to point directly at sun
2	HIRISE	2520 lbs.	1300 lbs. 750 lbs. 330 lbs. 90 lbs. 50 lbs. (25 ea)	Optics-6' dia. x 10' Cryogen 3' dia. x 3' Data Processor 3' x 2' x 2' Storage 1' x 1' x 1' (2) Control Boxes 2' x 1' x 1' (Loc. in S/C Cabin)	• 4-6 Sortie flights of 7-30 days each intervals between each flight to be several months • Experiment to be oper. 10% of time on orbit. Not uniformly spaced	Specialist required to p point optics, select targets, focus sensor, and edit data	• 28 ± 4 Volts • 420 Watts • 45 Whr per revolution • 10 Watts stand-by • ~5.04 kWhr max.	Approx. 400W dissipation by radiator	None	• Optics pointed within 50 NM on either side of ground track & stabilized in all 3 axes during acquisition • 0.5° ± 0.1° along optical ref. line prior to acquisition. Less than 2 arc. min. during acquisition for alignment • Less than 6 arc. min./sec. jitter rate of optical axis during acquisition for stability		• A precision pointing instrument such as the LRC annular magnetic suspension may be required. Optics to be designed to interface with LRC system
3	Atmospheric Topside Laser Sounder (ATLAS)	340 lbs.	20 lbs. 130 lbs. 40 lbs. 150 lbs.	Transmitter 36" x 8" x 8" Receiver 20" dia. x 60" YOKE Power Supply 2' x 2' x 4'	• 7-14 days • Data taken during each nighttime orbit	Specialist to turn experiment on and off and adjust the pulse repetition rates	• 28 ± 4 Volts • 600 Watts for 1/2 PPS laser converter • 300 Watts for 1/4 PPS laser converter at 450 Hz • 100 Watts for electronics	• Laser rod temperature maintained below 30°C during operations	• All data to be recorded on tape recorder and retrieved upon return of orbiter • Voice communications required with ground to discuss experiment status	• Orbiter required to point laser beam at nadir to within a few degrees (2) for alignment • ± .5 degree for angle determination for knowledge	N/A	

										rate of optical axis during acquisition for stability		
3	Atmospheric Topside Laser Sounder (ATLAS)	340 lbs.	20 lbs. 130 lbs. 40 lbs. 150 lbs.	Transmitter 36" x 8" x 8" Receiver 20" dia. x 60" YOKE Power Supply 2' x 2' x 4'	• 7-14 days • Data taken during each nighttime orbit	Specialist to turn experiment on and off and adjust the pulse repetition rates	• 28 ± 4 Volts • 600 Watts for 1/2 PPS laser converter • 300 Watts for 1/4 PPS laser converter at 450 Hz • 100 Watts for electronics • 37.8 kWhr max.	• Laser rod temperature maintained below 30°C during operations	• All data to be recorded on tape recorder and retrieved upon return of orbiter • Voice communications required with ground to discuss experiment status	• Orbiter required to point laser beam at nadir to within a few degrees (2) for alignment • ± .5 degree for angle determination for knowledge	N/A	
4	Precision Release Accuracy Test	500 lbs.	• 500 lbs. • Does not include orbiter mounting and support hardware	• 10 release mechanisms and 10 release objects • Release objects will be conical or cylindrical 20" dia. x 35" L • Control console & photo equipment will be loc. in S/L or shuttle cabin.	• 2 hrs. set up time for each test. Data to be recorded after release for 1/2 hr. for each release • Up to 10 releases during 7-14 day mission	• Specialist to monitor prerelease conditions, execute actual release, and monitor trajectory data and direct high speed cameras • 2-1/2 hrs per release	• 28 ± 4 Volts dc • 300 Watts avg. for each release	N/A	• 8 discrete functions for each of 10 releases • 6 kbps data rate from each release • 6 kbps data rate for orbiter data recording for each release • Release objects & trajectories will be recorded by means of telemetered onboard attitude & acceleration sensors as well as high speed cameras from the orbiter • 10,300 bits per rel. or 108,000 max. for commands • 108,000 bits max. for data storage on orb. • Telemetry data for release objects to be received by orbiter and recorded for eventual return to ground	• Mechanisms controlled to ±0.5 deg. all axes during release operations • Angular rates prior to release shall be less than ±0.005 deg. per second • Known orientation to be ±0.1 deg. all axes during release and data record • Known angular rates of release mechanism shall be ±0.001 deg. per second	N/A	• velocity of released object relative to release mechanism following release will be less than 1 Fps • Subsequent recovery of object by orbiter may be required
5	Earth Limb Measurements Sensor	675 lbs.	• 212 lbs. • 5 lbs. • 8 lbs. • 450 lbs.	• 1 telescope cylinder 20" dia. x 55" L • 2 electronic boxes (1) 7x5x5.5" (2) 6x6x11.6" • 2 spherical cryogenic tanks 30" dia. • Control console will be in shuttle cabin	• Continuously throughout 7-14 mission	• Specialist to monitor exp. operations 1-2 hrs. a day	• 28 ± 4 Volts dc. • 170 ± 25W • ~28.6 kWhr max.	• Heat load of an internally produced 9 Watts plus the heat load of the aperture, which varies with variant input Flux.	• 15 discrete functions • 8 magnitude functions of 8 bits each • Scanning rates and limits must be provided as support functions derived from orbiter position and attitude real-time information, to provide a smooth scanning motion of the sensor	• Optical ref. axis of P/L to be maint. with 0.4 deg. • Jitter ampl. to be 0.033 degrees less than 0.01 deg., 1σ 1σ is desired • ±0.023 deg. knowledge in the vert. scan rotational dir. & ±0.15 deg. in the other rotational axes • Scanning req'd in the region of the horizon (limb) to the local horizon with the scanning motion parallel		• Sensor to be mounted on a gimbal system provided as part of HARDARE ON STR along with tanks

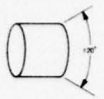
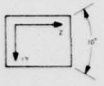
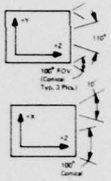
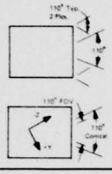
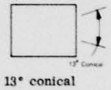
									• Telemetry data for release objects to be received by orbiter and recorded for eventual return to ground			
5	Earth Limb Measurements Sensor	675 lbs.	• 212 lbs. • 5 lbs. • 8 lbs. • 450 lbs.	• 1 telescope cylinder 20" dia. x 55" L • 2 electronic boxes (1) 7 x 5 x 5.5" (2) 6 x 6 x 11.6" • 2 spherical cryogenic tanks 30" dia. • Control console will be in shuttle cabin	• Continuously throughout 7-14 mission	• Specialist to monitor exp. operations 1-2 hrs. a day	• 28 ± 4 Volts dc. • 170 ± 25W • ~28.6 kWhr max.	• Heat load of an internally produced 9 Watts plus the heat load of the aperture, which varies with variant input Flux.	• 15 discrete functions • 8 magnitude functions of 8 bits each • Scanning rates and limits must be provided as support functions derived from orbiter position and attitude real-time information, to provide a smooth scanning motion of the sensor	• Optical ref. axis of P/L to be maint. with 0.4 deg. • Jitter ampl. to be 0.033 degrees less than 0.01 deg., 1σ 1σ is desired • ±0.023 deg. knowledge in the vert. scan rotational dir. & ±0.15 deg. in the other rotational axes • Scanning req'd in the region of the horizon (limb) to the local horizon with the scanning motion parallel to the local vertical in the orbital plane as well as off the plane up to ±30°		• Sensor to be mounted on a gimbal system provided as part of HARDARE ON STR along with tanks
6	Low Altitude Satellite Study of Ionospheric Irregularities (LASS II)	"A"-116	"A"-.40 lbs. .12 lbs. .44 lbs. .20 lbs.	"A"-Receiver • Transmitter • p^3 experiment 28" x 2.5 x 1.2" H • Electronics 7.5"x 7.0"x 6.48" H • FAR UV Airglow Experiment (Optional) 19" x 8.8" x 5.9" dia. • Tape recorder • Motor driven 30 ft extendable boom	• Maximum duty cycle will be 33% • A & B will be on all night	• Specialist to extend and retract the 30 ft. boom	"A"-40 Watts ~2.24 kWhr max.	N/A	• 4-frequencies radio beacon • All commands to be sent directly from ground • "A" power on/off for 3 experiments, tape-on-and-off & 6 undefined words for p^3 • 512 kb/s of nominal capacity for data from "A" and "B"			• p^3 and magnetometer booms will be folded • Overall deployed height is 20 ft. and 6-1/2 ft. dia.
		"B"-522	"B"-100 lbs. 50 lbs. 25 lbs. 132 lbs. 122 lbs. 15 lbs. 10 lbs. 68 lbs.	"B"-S/C Housing • ACS • Fuel (hyd. and thrusters) • Experiments • Power Supply • TLM, Cabling S Band, VHF • Transmitter • Safety (15%)		"B" Specialist to deploy and retrieve satellite	"B"-115 Watts	N/A	• Recorder dump once per orbit at 1 Mb/s • "B" pwr. on-and-off for 8 exp's, several 1 time funct, and 12 undefined words for exp. oper. modes. Max. word size is 32 bits. • Tape recorder of nominal capacity	"B" Spin vector perpendicular to orbit plane. Exps. in despun section must look into the ram vector to within 1° or better along 2 orthogonal axes. ± 1/2° for despun sect. along both axes.		"B" has a spinning sect. spun within orbiter and ejected at a V (Horz.) of 1/2 FPS. Recovery required.

Table 3-2. Experiment Requirements (cont.)

No.	Title	Total Weight	Weight Break Down	Size	Duty Cycle	Astronaut/ Specialist Participation	Experiment Power	Thermal	communications	ACS	FOV	Special
7	CRLS-262 XUV Environment in Space (4 modules)				TBD	N/A	• 28 ± 4 Volts dc.	• Non-oper. surviv. upper limit set by evap. of photocathode materials (50°c)	• Plus timing & sync. • No serial comm. required. May be reduced by auto seq. into exp.	• Accurate point. & ephemeris data required		• Exp. is passive except for occasional calibration or filter chg. • Outgassing rate from detectors is 1.5×10^{-4} gms of isobutane per day max. • Moving mechanisms may have to be caged during handling & shake tests on ground • Exp. designed to fly on the pallets of a 3 axis stab. nadir oriented host vehicle
	Zenith sky monitor	30 lbs.	30 lbs.	10 x 10 x 25"			• 5 Watts • ~ 4.03 kWhr max.	• 10 to 20°c preferred -20 to 35°c acceptable	• 25 kbps 4 commands	• Rel. align. knowledge is ±0.1° • Rel. align. know. to satellite ACS to be ±0.25° in pitch and yaw, otherwise point. ref. must be obtained from internal star tracking capability. • 9 sensors meas. range 5°A to 80°A		
	Auroral Monitor	25 lbs.	25 lbs.	8 x 12 x 12"			• 5 Watts	• 10 to 20°c preferred -20 to 35°c acceptable	• 2.5 kbps 8 commands	• 1 sensor 5Å to 80Å 2 sensors 100Å to 600Å 1 sensor 5577°c		
	All sky monitor (+ Y & -Y pallet)	50 lbs.	25 lbs. each	12 x 12 x 15" each			• 14 Watts (7W each)	• 10 to 20°c preferred -20 to 35°c acceptable	• 5 kbps (2.5 kbps ea.) 12 commands (6 each)	• 1 ea. - Y & + Y sensor 1A to 16Å 2 ea. - Y & + Y 100Å to 600Å		
8	Optical Geophysics and Astronomical Observatory	120 lbs.	120 lbs.	30 x 48 x 15" ultraviolet image converter camera	• 10% operation whenever suitable regions of interest are available • 1 hour per exp. viewing period	• Specialist will manually control pointing of experiment, select and monitor regions of interest and control viewing to achieve time histories	• 28 ± 4 Volts dc. • 1.68 kWhr max.	N/A	• Voice comm. with ground • 1.0 Mbps real-time slow video link to prechosen grd. suppt. sta. • 1.3 kbps of sensor data relayed in real-time to grd. sta. or stored for eventual return to grd.	• Optical axes ref. line to be within ±1.0° during OPS for align. & knowledge • Allowable jitter rate of the optical axes is less than 0.005 deg. per second	 13° conical Optical axes parallel to largest dimension	• May be located in a sharing mode on an azimuth-elevation gimbal system platform provided as supporting hardware • Camera usual central view position to be directed at the nadir.
9	Teal Ruby	724 lbs. (max)	531 lbs.	53" x 44" x 93"	• 1 Oper=20 min. • Max. # of oper. per 1 orbit = 2 • Max. # of opers. per 14 orbits = 2	Specialist initiates sys. operations. 3 are required for 24 hour coverage for 448 individual trackings	• 28 ± 6 Volts dc • 312 Watts avg. • 422 Watts peak • 50 Watts	• External temp. of exp. to be 290 ± 5 degrees kelvin • Zero heat transfer between	• High rate data sensor output is serial digital rate of 1024 Kbps in real-time • Housekeeping digital data is 64 eight-	• Ephemerides position accuracy reqmts: In Track Days 0 1	• In all oper. positions a clear FOV 1-1/4° in all directions from sensor line of sight • Pitch motion -60° to + 80° from nominal position with line of sight at 0° to nadir axis	• Potential hazards to personnel Liquid Helium -255°C Solid Neon -259°C Solid Methane -233°C • Cover assy. soft latching device uses permanent magnet of less than 0.3

9	Teal Ruby	724 lbs. (max)	531 lbs.	53" x 44" x 93"	• 1 Oper=20 min. • Max. # of oper. per 1 orbit = 2 • Max. # of opers. per 14 orbits = 2 • Max. overall oper. duty cycle = .8%	Specialist initiates sys. operations. 3 are required for 24 hour coverage for 448 individual trackings	• 28 ± 6 Volts dc • 312 Watts avg. • 422 Watts peak • 50 Watts standby for 19 hrs. (max) prior to oper. • 15 Watts continuously for duration of mission • ~10 kWhr max.	• External temp. of exp. to be 290 ± 5 degrees kelvin • Zero heat transfer between exp. & S/C is goal • Exp temp. to be within -20° to +40°C during prelaunch	• High rate data sensor output is serial digital rate of 1024 Kbps in real-time • Housekeeping digital data is 64 eight-bit digital sample once/sec • Fast memory dump is digital words of 32 bits numbering 800* • Data record/store for 1 oper = 1.2×10^9 bits. Playback rate to be 1024 Kbps • S/C to be capable of storing 256 16 bit magnitude control words transfer rate to exp. to be 1000 wps using a 3 bit rate clock at 64 Kbps • Binary VTCW at 512 Kbps length to be 26 bits • 21 discrete on-off commands • 2 discrete on-off signals from exp.	• Ephemerides position accuracy reqmts: In Track Days: 0 — 1; 900' — 3900' Cross-Track: 0 — 1; 100' — 100' Alt.: 0 — 1; 100' — 100' • After acquisition target to be tracked for 12 minutes. Maximum of 4 targets per orbit data to be dumped during subsequent orbit	• In all oper. positions a clear FOV 1-1/4* in all directions from sensor line of sight • Pitch motion -60* to +80* from nominal position with line of sight at 0* to nadir axis • Roll motion ±10* from nominal pos.	• Potential hazards to personnel Liquid Helium -255°C Solid Neon -259°C Solid Methane -233°C • Cover assy. soft latching device uses permanent magnet of less than 0.3 Gauss at radial distance of 15 cm along any axis Gimbals have less than 0.1 Gauss at a radial distance of 15 cm along any axis
10	Physics International Modular Bremstrahlung Source (PIMBS II) Experiments	B^1- 27,836 B^2- 30,453	SOURCE- 14,636 lbs. X-ray generator 8,000 lbs. (included in source) gas enclosure 11,400 lbs. STO- 1,000 lbs. IUS cradle 700 lbs. other ASE 100 lbs. same as B^1 above, 2nd RMS 867 lbs. storage container 1750 lbs.	B^1-8 x 6-1/2 x 10' H Same as B^1 above plus: 600" Long to accommodate 11-3 cubic ft. es. STO's	1 STO. It must be rotated sequentially to expose each side to the source 11 STO's each must be pulsed 9 times at various orientations in relation to source	Specialist will be required to operate RMS, deploy source & PTO and retract & store same Same as B^1 above	• 2 kW sub. sys. 3 kW for 45 sec. charging power • 5×10^6 amps current to generate X-rays • batteries or S/C power use to be determined • 5.5 kWhr max. Same as B^1 above	N/A	N/R 250 BPS-64 kbps, sensors, PCM's, RCDR's, Switch matrix, and PSK SCO 1.024 MHz (28 Vdc)	N/A	N/A	• 40 pulses per experiment configuration 100 per mission • 100 psig, 100^3 in SF6 vented every pulse • 140 psig, 3600^3 in SF6 (15%) argon (85%) vented every to pulses • Deionized, degassified H_2O circulation every cycle • 150 joule, 2π steradian output source plane • No RF, IR, visible or UV through screen • X-ray spectrum • Collect all vent. gas • Store SF6 as liq. at 310 Psia • Store SF6 argon as gas at 1K Psia • 20% residual for SF6 • Total quant. SF6 argon-3 times that req. to be vented to ret. pos. diff. for blow down to vent gas storage tank
11	Optical	50kg		70 x 70 x 50 CM^3	• 10 encounters	Specialist	• 28 Vdc	N/A	• 10 momentary	N/A		• Circular 400km orbit

			1750 lbs.	es. STO's	source							• Total quan. STO argon 3 times that req. to be vented to ret. pos. diff. for blow down to vent gas storage tank
11	Optical Counter Measures Demonstration	50kg 9kg	 3kg ea.	70 x 70 x 50 CM^3 radiometers (3) 12.5 x 15 x 17.5 CM^3	• 10 encounters minimum with the 2 test sites MIT & HAFB • each encounter to last 3-5 min. • max. No. of encounters per day is 4	Specialist will be required to operate RMS, deploy experiment and retract and store same	• 28 Vdc • 50 Watts • 0.05 kWhr max.	N/A	• 10 momentary commands • 40 digital 1 sec. -1 channels • 5 10 kHz wide band analog. channels	N/A		• Circular 400km orbit at 45° incl. polar or near polar incl. acceptable • Ascending node, incl. & period should optimize No. of high elev. passes over sites • Flts. - should be conducted in late summer or fall • 15 meter separation from orbiter • Short, 2 to 3M, retractable booms may be deployed
12	Deployable Spacecraft	300 lbs.		o 16^3 ft. for experiments o spacecraft 17" x 49" x 63"	• Experiment must be capable of 6 months storage • 194 days • Lew duty cycle updates up to 2 weeks apart	N/A	28 ± 4 Volts supplied by solar/array batteries 180 w hrs. (6.5×10^5 joules) for 1 orbit ea. day or 600w hr. (2.16×10^5 joules) in 1 day on alternating days	Internally generated heat loss to space provided by 3 thermal control louvers 60° to ±20°f (16 ± 11°c)	• Transmitter SGLS/PCM/PSK B1-Phase LPCM 20 watts at 2200-2300 MHz 16,384 bps on 1.024 MHz subcarrier 131,072 bps on 1.7 MHz subcarrier. 8 bits word length and 2048 words, total record is 2.06×10^8 bits • Data storage record time 210 minutes, playback compression 8:1, 2 tracks • Clock capacity 194 days, 61 sec. resolution 1 part in 10^5 accuracy • 128 on-off commands • 8 bit magnitude commands • 3 magnitude & 60 on-off commands available to experimenters	• Programmable spin & precession electromagnets to main. spin rate and attitude with a passive wobbler damper commands generated from ground using real time sensor data • Earth sensor, sun sensor, and magnetometer are used singly or in combination to provide redundant attitude determination data • Spin rates to 12 RPM • Spin axis to within 5° of normal orbit • Spin rate to within 0.05 RPM/Day • Maneuver rates to 10 deg./orbit • Spin rate changes to 20 RPM/Day • 3 axis attitude determination from single station pass • Attitude determination to 0.5 to 1.5 deg. all axes • Ephemeris determination to ±0.5 to 1.5 • Spin rate determination to better than 0.01 RPM	N/A	• Electro explosive devices must be selected from approved list • Grounding & bonding PERMIL-B-5087B • Electromagnetic compatibility per MIL-STD-461A • Apogee altitudes to 5000 NM (9300 km) • Deployable free flyer

3.3 SPACE TRANSPORTATION SYSTEM (STS) IMPACT

Ten of the candidate experiments may use some degree of Mission Payload Specialist participation either before, during, or after the initiation of the experiment and on occasion in all of the phases. This participation ranges from minor monitoring operations to more complex tasks of using the Shuttle booms to select experiment objects, extend them beyond the shuttle bay, and perform numerous orientations for gathering of experiment data. Significant integration with the STS Project Office will be required to integrate the experiment requirements for the participation by the Mission Payload Specialist and the integration and installation of required control panels and equipment.

The need for specific Payload Specialist participation must be examined and carefully evaluated as it has significant impact on autonomy and quick reaction capability. Other STS considerations which impact the STR concept are as follows:

1. Acceptability of RF in the cargo bay area must be determined as to the allowable levels, mode of integration with other equipment, etc. The assumed guideline is that RF in the cargo bay is acceptable.

2. The STS caution and warning requirements will be minimized by the use of high margins for STR equipment. The requirements due to experiments must be evaluated and negotiated with the STS Project Office.

3. Current STS requirements are that commands to the payloads go through the crew and not directly from ground or user to payload. This requires a STR/STS interface for command override.

3.4 PLANNING IMPACT

The simplicity of the STR systems and subsystems minimizes the need for complex integration planning in that the majority of the experiments and the STR system will be available for predetermined Shuttle flights and as a goal, should be installed, tested, and verified for flight within five days of launch. The most critical planning effort will involve tracking of the Shuttle schedule so that the STR/Payload can be responsive to flight opportunities. This involves integration of Mission Payload Specialist requirements and the integration and installation of associated equipment.

The dates must be established with NASA as to the latest possible time prior to launch to accomplish the following STS integration activities to allow the hardware interfaces to be carried out.

1. Analytical and design interfaces definition
2. Preinstallation acceptance of STR payloads

3.5 OPERATIONS IMPACT

The STR system and subsystems are designed to minimize the amount of test and installation time required to integrate the STR equipment on board the Shuttle. Each subsystem can be pre-assembled on equipment panels that are designed to fit interchangeably on the STR configurations. Each STR/equipment assembly will have end-to-end checkout and acceptance testing performed prior to installation in STS, thereby minimizing STR/STS on-line integration time. This will permit symmetry of STR installations, thereby permitting the most advantageous locations to satisfy the center of gravity needs of the Shuttle.

3.6 SUMMARY OF MAXIMUM STR DESIGN REQUIREMENTS

Figure 3-1 shows the results of the requirements synthesis of the twelve experiments and represents the maximum requirements for designing the STR modular subsystems.

SIZE	
HIRISE EXPERIMENT	OPTICS - 6' DIA. X 10' LONG CRYOGEN TANK - 3' DIA. X 3' LONG DATA PROCESSOR - 3' X 2' X 2' STORAGE MODULE - 1' X 1' X 1' CONTROL BOXES (2) - 2' X 1' X 1' IN S/C
POWER 28 ± 4 VOLTS dc	
STEADY STATE (ATLAS)	600 Watts
CONTINUOUS (ELMS)	170 Watts
PEAK (PIMBS II)	3.0 kW FOR 45 SECONDS
TOTAL ENERGY (ATLAS)	37.8 kWh
THERMAL	
NOMINAL TEMPERATURE RANGE (ALL)	0 - 30°C
DISSIPATION (STEADY STATE) (PIMBS)	0 - 1 kW (3 kW PEAK FOR 45 SECONDS)
COMMUNICATIONS	
COMMANDS (TEAL RUBY	1 Kbps
DATA PROCESSING (TEAL RUBY)	1024 Kbps IN REAL TIME
RECORDING (LASERCOM)	*1 Mbps*
ATTITUDE CONTROL	
KNOWLEDGE (PRAT)	POINTING - 0.1° (.023° VERTICAL SCAN) JITTER RATE - 0.001°/SEC.
POINTING (ELMS)	IN ORBITAL PLANE, FROM HORIZON TO NADIR TO HORIZON ±30° OUT OF ORBITAL PLANE
STABILITY (TEAL RUBY)	0.01°/SEC. 0.004°/SEC. JITTER RATE
SPECIAL ITEMS	
FIELD OF VIEW (CRLS-262)	THREE AXIS STABILIZED ON A NADIR ORIENTED HOST VEHICLE USING FOUR DETECTION SYSTEM MODULES. ONE WITH THE CENTER OF ITS FIELD OF VIEW (FOV) POINTED TOWARD THE ZENITH (-Z) AXIS, ONE ON THE +Y SIDE AND ONE ON THE -Y SIDE EACH WITH THE CENTER OF THEIR FOV IN THE Y, Z PLANE 45° BELOW THE -Z, ONE WITH THE CENTER OF THE FOV DIRECTED AT NADIR. NOTE: ASSUMED COORDINATE SYSTEM IS: +X VELOCITY VECTOR +Z NADIR +Y RIGHT HAND RULE
PAYLOAD SPECIALIST NEED	**10 OF THE 12 EXPERIMENTS MAY USE SOME DEGREE OF PAYLOAD OR MISSION SPECIALIST PARTICIPATION EITHER BEFORE, DURING, OR AFTER THE INITIATION OF THE EXPERIMENT AND ON OCCASION IN ALL OF THE PHASES AS NOTED BELOW**
NUMBER OF EXPERIMENTS	*DEDICATION – 10 HOURS A DAY = 100%*
1 (TEAL RUBY)	100% - 3 SHIFT
5	30 TO 50%
2	10 TO 20%
1	TURNON-TURNOFF 10%
1	RMS - DEPLOY & RETRIEVE 1/2 WORKING PERIOD EACH
2	NONE

EXPERIMENT NO. 10 (PHYSICS INTERNATIONAL MODULAR BREMSTRAHLUNG SOURCE (PIMBS II) IS A SPECIAL CASE BECAUSE OF ITS LARGE WEIGHT AND SIZE AND IS NOT INCLUDED IN THESE REQUIREMENTS EXCEPT FOR ITS PEAK POWER REQUIREMENT.

Figure 3-1. Maximum STR Design Requirements

SECTION 4

STR PRELIMINARY DESIGN (TASK 2)

The design of the STR was based on the requirements synthesis of the twelve experiments as described in Section 3 of this report (Task 1). In addition, minimum life cycle cost was a major driver.

As a result of subsystem design based on these requirements, the STR is a system of standardized modular elements (structural and functional), designed for mission flexibility, ease of shuttle integration and low cost.

The STR is standard in that it can fly on most flights (more than 30 per year) and has a minimum standard interface with shuttle. It is also selectable in that it consists of an "Erector Set" of modular components which can be rapidly assembled to meet a variety of mission opportunities. The primary objectives of STR are to:

1. Provide the DOD R&D community maximum opportunities for spaceflight
2. Minimize spaceflight cost in shuttle era

STR helps achievement of these twin objectives by:

1. Development of low cost experiment support system/subsystems for shuttle flight
2. Use of minimum Orbiter interfaces
3. Use of Orbiter Bay excess capabilities
4. Development of standard experiment interfaces
5. Use of proven hardware
6. Use of modular approach

Some of the salient design features of the Standard Test Rack are:

1. Autonomy
   a. Power, attitude knowledge, thermal control
   b. Communications
   c. Attitude Pointing (Points)
   d. Quick reaction capability
   e. Security
2. Modularity
   a. Equipment panels interchangeable
   b. Battery Power
   c. Change of phase thermal control
   d. Multiple configurations - strongback, adjustable bridge, shelves, POINTS
3. Ease of Integration
   a. Install within 5 days of launch
   b. Standard Interfaces
   c. Minimum shuttle interfaces
4. Life Cycle Costs
   a. High margins
   b. High reliability components
   c. Workhorse approach
   d. Cost effective over wide range of payload size
   e. Short length (4') $\Rightarrow$ more available flights
   f. Uses excess space (around tunnel)
   g. Ease of maintenance/refurbishment/reuse

Figure 4-1 shows the overall STR design concept. Detail drawings are provided in Appendix E so that they may be viewed while reading the text. The STR is a complete system within itself (See Figure 1-5) that can be flown on any STS flight cargo configuration of spacelab modules and pallets, spacecraft and spacecraft on upper stages (IUS, SSUS). The STR is designed as a secondary payload that will carry payloads being tested to verify their performance for operational use, R&D operation missions, and operational user missions.

Complete autonomy of the STR with respect to the STS is limited by:

1. Safety of the Orbiter in flight
2. Astronaut participation required to operate the payloads

A summary of the autonomy issues is shown in Figure 4-2.

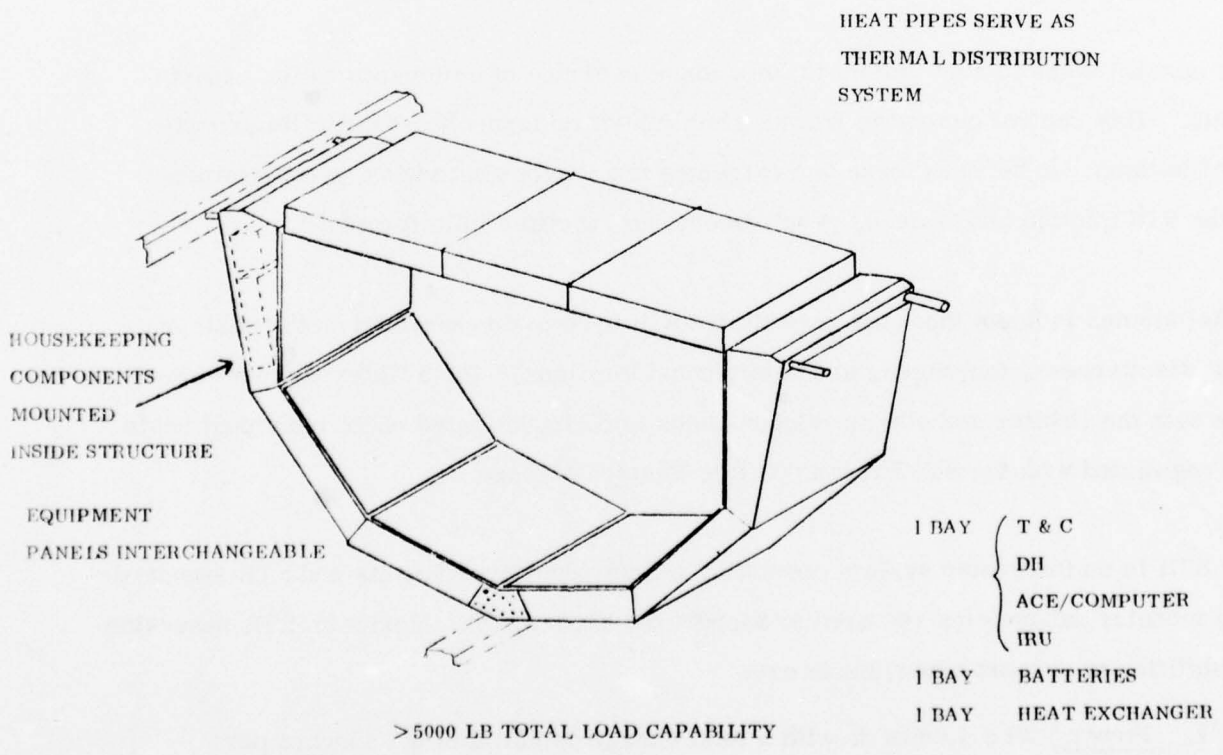


Figure 4-1. Design Concept

Shuttle Services	Autonomy	Orbiter Restraint
• Structural	X	
• Thermal	X	
• Power	X	
• Communication		
- Command		X
- Telemetry	X	
• Attitude Control		
- Position		X
- Knowledge	X	
- Pointing	X	
• Crew Participation		X
• Safety		X
• Command	- Crew Override - MCC Override	
• Attitude Control Position	- Overall Mission Limits	
• Crew Participation	- As Required	
• Safety	- Caution & Warning - Jettison Requirements - Turn-Off - Emergency Procedure	

Figure 4-2. STR Autonomy Issues

The current trend in crew safety is for manual override of commands to the experiments. This control can range from a simple turn-on turn-off capability to experiment jettison. In between these two extremes are various activities as determined by the STS Caution and Warning requirements as specified in Reference 1.

Initial studies indicate that each experiment will have to be examined on the basis of cost effectiveness, functional, and safety considerations. The STR/experiment interface with the Orbiter and other payloads needs to be investigated on an individual basis and negotiated with the STS Program Office Mission Managers.

The STR is an integrated system consisting of individual experiments and STR standardized modular subsystems required to support the experiment. Maximum STR subsystem capabilities to support experiments are:

1. Power. 28 ± 4 volts dc with a total energy potential of 37.8 kwhrs per battery module.

2. Thermal. 0-30° centigrade with 0-1 kW dissipation (steady state) range.

3. Communications. 1 Kbps for commands, 1,024 kbps for real-time data processing, and 1 Mbps for data recording.

4. Pointing. In orbital plane, from horizon to nadir to horizon, ±30° out of orbital plane.

5. Structure. Capable of supporting >5000 lbs for entire STR system.

The key features of STR are:

1. The modular construction of four basic STR configurations shown below that will be compatible with the primary shuttle payloads.

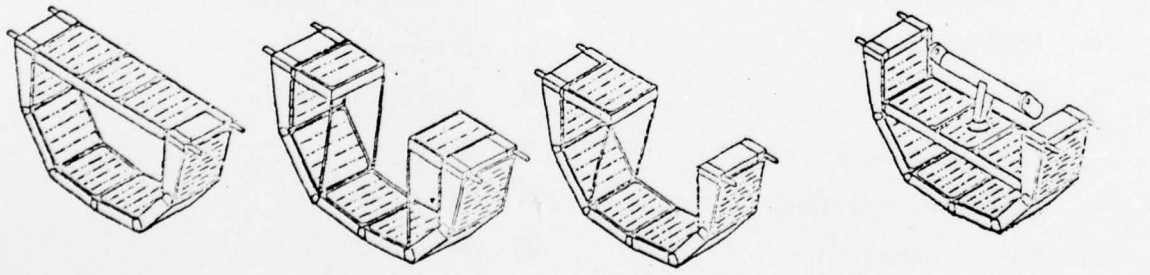
   a. Bridge

   b. Dual Shelf

   c. Single Shelf

   d. Bridge with Gimbal

(See also Figures 4-3 through 4-13)



BRIDGE DUAL SHELF SINGLE SHELF BRIDGE WITH GIMBAL

2. STR provides flight opportunity for experiments or sensors with minimum interface requirements and quick reaction to flight demand.

3. Structured flexibility is a key STR characteristic which permits the efficient use of unfilled shuttle payload bay space on nearly all shuttle flights.

4. STR Support Subsystems are available as building blocks to permit a wide range of experiment or operational systems support capability.

5. A simple, minimum interface with the shuttle orbiter minimizes the integration time, complexity and cost of STR integration.

The STR subsystems will provide attitude and structural support, control the STR system temperatures, furnish electrical power, provide pointing and tracking data, and manage, process, and store the command and telemetry data. The major elements of each subsystem are:

1. Structures
   a. Strong back frames
   b. Trunnions
   c. Closure panels
   d. Interchangeable Equipment panels
   e. Shelves
   f. Supports for POINTS and Payloads

2. Thermal
   a. Heat pipes
   b. Thermal blankets
   c. Heat exchanger

3. Power
   a. $LiSO_2$ Batteries
   b. Ordnance controller
   c. Power controller
   d. Wire harnesses

4. Pointing and Tracking
   a. POINTS system (Payload Orientation and Instrument Tracker for Shuttle)
   b. Sun Sensors (2)
   c. Star Trackers (2)
   d. NSSC-1 computer
   e. IRU

5. Communications
   a. Antenna (2 turnstiles)
   b. Antenna (1 hybrid)
   c. Diplexer
   d. Transmitter
   e. Receiver
   f. Recorder
   g. Telemetry Unit
   h. Decrypter
   i. Encrypter
   j. Command Decoder
   k. Data processor

Table 4-1 presents the STR/Experiment weight summary for each of the twelve experiments. Figures 4-3 to 4-13 show sketches of each experiment integrated on the STR. Drawings STR004 to STR011 in Appendix E provide additional detail information.

4.1 ELECTRICAL POWER SUBSYSTEM

The STR provides Electrical System Services to the various payloads and to the STR subsystems as required to accommodate the power, pointing, command, and telemetry requirements for a given mission.

4.1.1 ELECTRICAL POWER REQUIREMENTS

Power requirements for the candidate experiments were analyzed to determine the type of power required to be supplied by the STR in support of the experiments. (See Table 4.1-1.) The analyses concluded that all of the candidate experiments require 28 ± 4 Volts dc. The maximum power required would be 600 Watts average daily for 6 out of 16 revolutions of the Orbiter per day with a 3 kW peak for 45 seconds. The maximum energy required for an experiment and a seven day mission is 37.8 kWh for Experiment No. 3. The experiments requirements are described in Section 3, Table 3-2 in this report. The maximum total energy requirement is 60.9 kWh for Experiment No. 5.

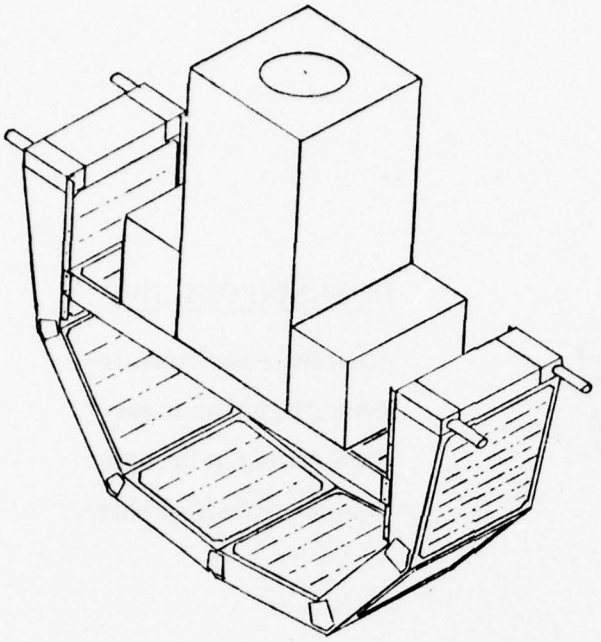
The support subsystems, Communications, Attitude Control and Electrical Power, power requirements are included in Table 4.1-1. Thermal power requirements to maintain a temperature of 0 to 30^{o}C for each subsystem are also included in Table 4.1-1.

4.1.2.1 Autonomous Power Source Selection and Design

Since the total power capability of the Shuttle is 7 kW and 50 kWh on a shared basis and previous studies indicate high usage rates, it will be necessary for the STR to provide an autonomous power source or an additional Shuttle power kit. The kit provides 840 kWh and is chargeable from a cost and launch weight standpoint to the payloads using it. The autonomous power source will provide the independence from priority and capability for last minute integration desired for the STR concept. Several methods of providing the additional power were investigated and a summary of the comparisons are contained in Tables 4.1-2 and 4.1-3.

Table 4-1. STR/Experiment Weight Summary

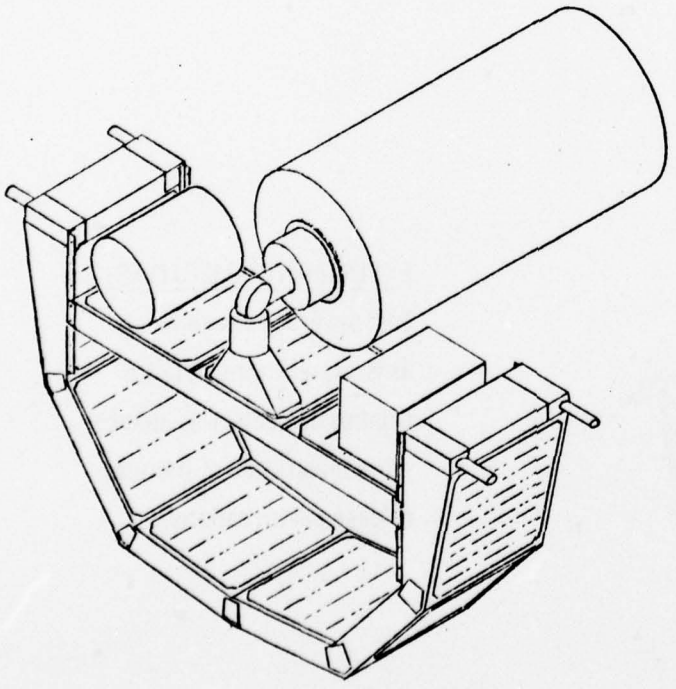
Subsystem	Experiment											
	Laser Com	HIRISE	Atlas	Prec. Rel.	Elms	Lass II	XUV	OGAO	Teal Ruby	Pimbs	OCD	DS
Structure	1480	1480	1480	1480	1480	1480	1480	1480	1105	Too heavy to be considered for STR	1480	1480
Communications	99	99	99	99	99	99	99	99	99		99	99
Power	130	130	130	130	130	130	130	130	130		130	130
Attitude Control	106	106	106	106	106	106	106	106	106		106	106
Thermal Control	220	220	220	220	220	220	220	220	185		220	220
Batteries	300	225	375	150	488	150	263	188	188		75	75
Heat Exchanger	888	444	888	888	888	888	888	444	444		444	444
Points	-	450	-	450	450	-	-	450	450		-	-
Experiment	1050	2520	340	500	675	638	53	120	724	27836	110	?
Total	4273	5674	3638	4023	4536	3711	3239	3237	3431	-	3664	2554



EXPERIMENT OBJECTIVE:

Evaluate Space to Space high data rate (1 Gbps) transmissions on an optical laser link.

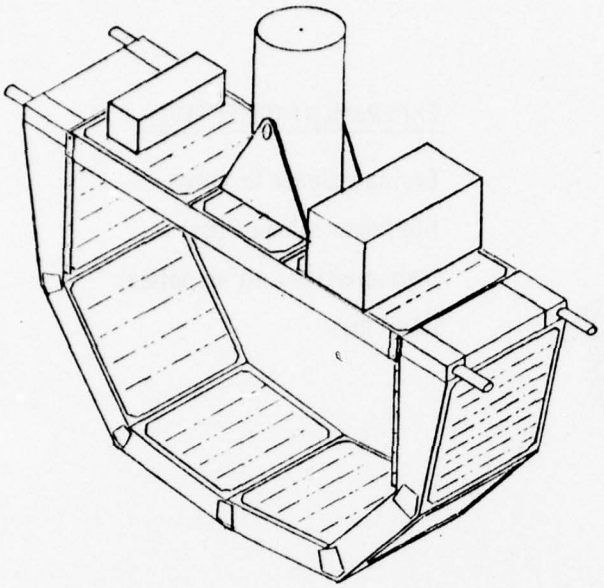
Figure 4-3. Lasercom Receiver Test



EXPERIMENT OBJECTIVE:

Characterize Geothermal sites on a global basis.

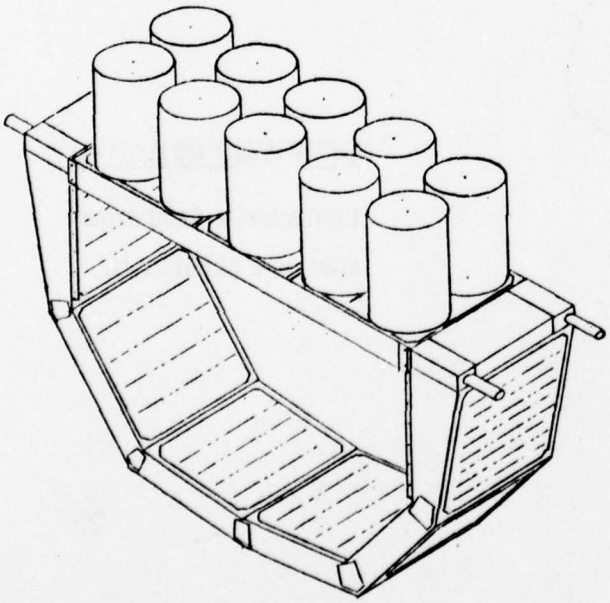
Figure 4-4. Hirise Experiment



EXPERIMENT OBJECTIVE:

Accurate remote measurement of atmospheric density & aerosol in the upper troposphere & stratosphere.

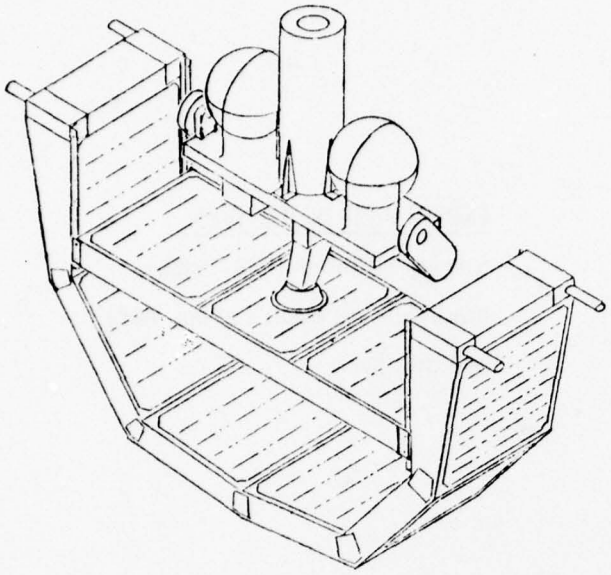
Figure 4-5. Atmospheric Topside Laser Sounder (Atlas)



EXPERIMENT OBJECTIVES:

Test several release mechanisms, examine detailed relative trajectories, determine magnitude of induced release perturbations.

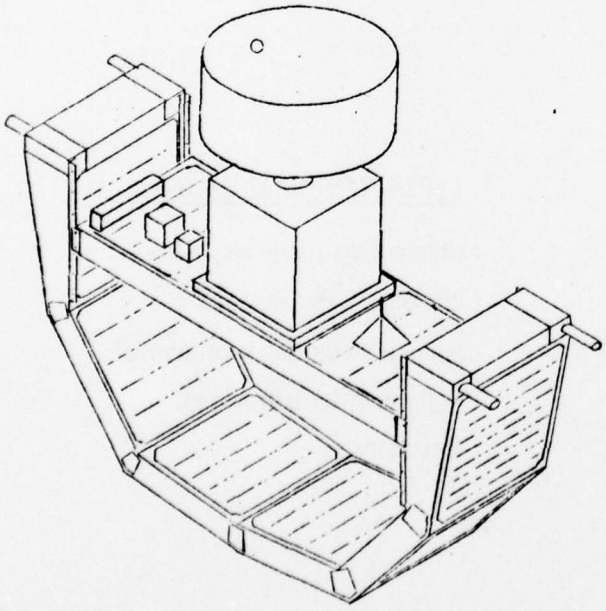
Figure 4-6. Precision Release Accuracy Test



EXPERIMENT OBJECTIVES:

IR telescope to provide earth limb radiance variance as function or altitude above the limb.

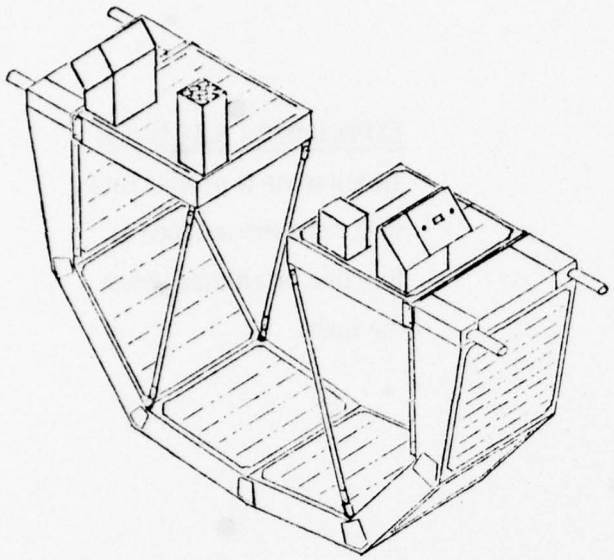
Figure 4-7. Earth Limb Measurements Sensor



EXPERIMENT OBJECTIVES:

Measure VHF/UHF scintillation phenomenon to understand cause/effect of plasma instabilities & ionospheric irregularities.

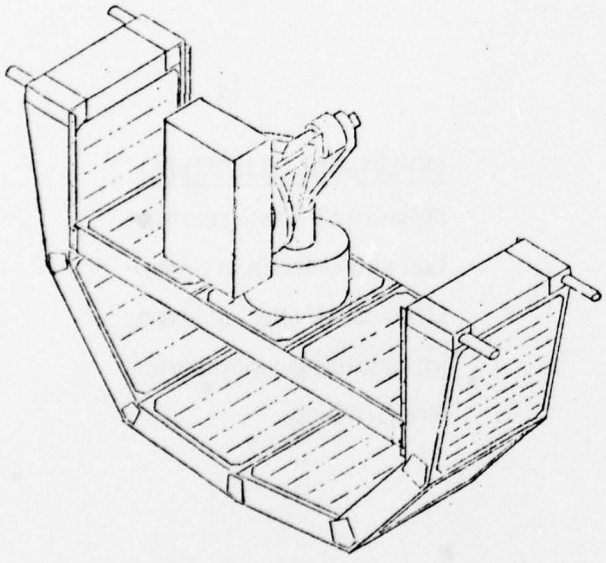
Figure 4-8. Low Altitude Satellite Study of Ionospheric Irregularities (Lass II)



EXPERIMENT OBJECTIVES:

Survey XUV and X-RAY background of the earth's atmosphere and the sky.

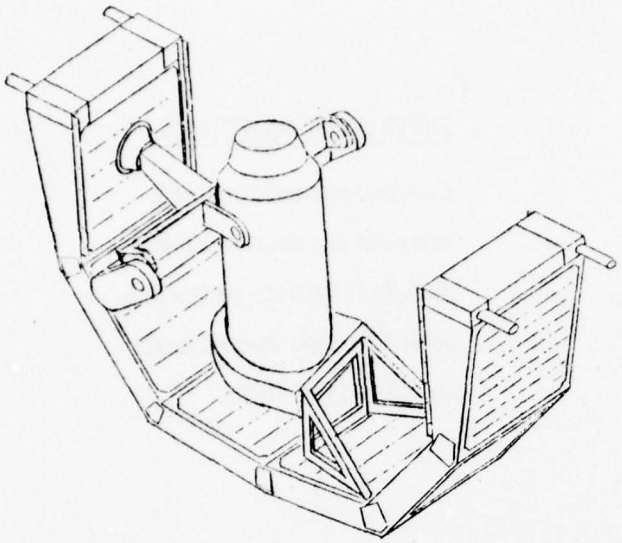
Figure 4-9. XUV Environment in Space



EXPERIMENT OBJECTIVES:

Obtain time histories, morphology, & dynamics of auroral activities & equatorial regions of far ultraviolet emissions.

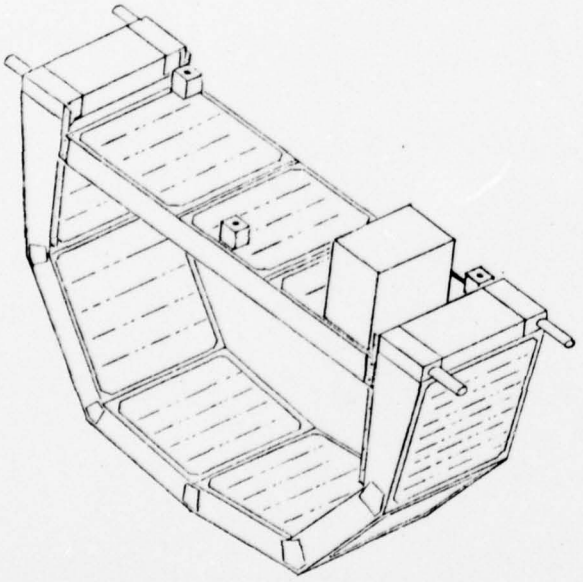
Figure 4-10. Optical Geophysics and Atronomical Observatory



EXPERIMENT OBJECTIVES:

Test cryogenically cooled mosaic focal plane of telescope which provides several spectral bandpasses.

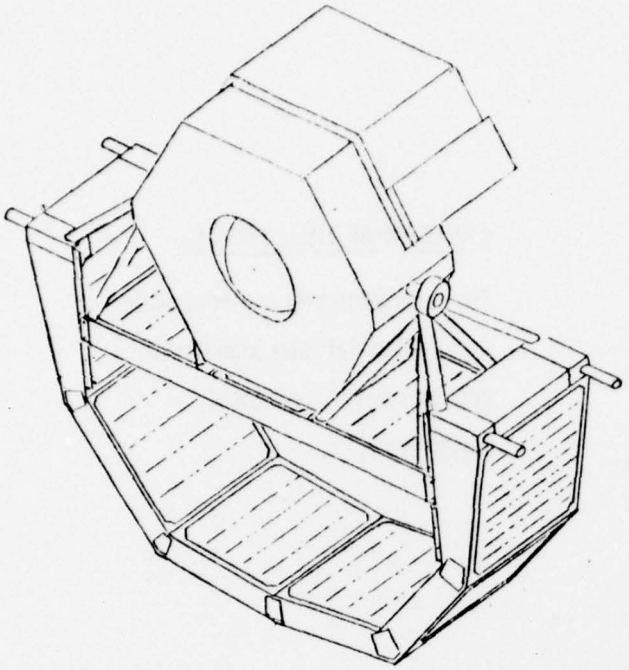
Figure 4-11. Teal Ruby



EXPERIMENT OBJECTIVES:

Demonstrate performance of optical countermeasures against lasers.

Figure 4-12. Optical Countermeasures Demonstration



EXPERIMENT OBJECTIVES:

Demonstrate deployment & retrieval of standard service module (SSM) for supporting scientific experiments & development test items.

Figure 4-13. Deployment Spacecraft

As shown in Table 4.1-2 seven types of aerospace power sources were considered. The comparisons shown indicate that use of primary batteries is the desired source of power for the STR system. Primary batteries provide the highest energy density which allows for a power module (panel) with the greatest total energy capability. Primary batteries are also the most cost effective approach for short length missions as intended for STR payloads.

Various primary battery types were evaluated and the comparisons are noted in Table 4.1-3, Battery Options. The Lithium Sulfur Dioxide battery was selected as the power source because of minimum size, weight and cost.

Table 4.1-1. Summary Power Requirements Power (Watts)

Exp No.	Voltage	Ave	Peak	Stand By	Total Energy, WH Experiment Only	Total
1	28±4V	-	1000	100	23,100	33,500
2	28±4V	45	420	10	5,040	26,900
3	28±4V	600	-	-	37,800	46,800
4	28±4V	300	-	-	7,500	18,400
5	28±4V	170±25	-	-	28,600	60,900
6	28±4V	115	-	-	2,240	18,700
7	28±4V	24	-	-	4,030	30,600
8	28±4V	-	-	-	1,680	20,300
9	28±4V	312	422	50	10,000	19,000
10	28±4V	2000	3000	-	5,500	Not Determined
11	28±4V	50	-	-	50	5,700
12	28±4V	30	-	-	600	8,600

Table 4.1-2. Power Subsystem Comparisons

	Autonomy	Modularity	Ease of Integration	Cost ($/Kilowatt-Hour)
Primary Batteries	Two Terminal Device, Cubic Packages	Complete Range of Sizes, Easily Paralleled	Simplest Energy Storage System	$1000 per Kilowatt-Hr, Up to 50% Recovery on Return
Secondary Batteries	Must be Recharged From Some Other Energy Source	Complete Range of Sizes, Easily Paralleled	Recharge Provisions Must Be Considered	$50,000 per Kilowatt-Hr Ten Year Life
Fuel Cell	Requires Cryogenic Tankage, Active Coolant System, and Purge and Vent Lines	Few Sizes Available, but Easily Paralleled	Cryogenic Storage Purge and Vent Lines, Fuel Tanks Topped on Pad	$1,800,000 per Kilowatt-Hr with 30 Day Limit on Cryogenic Storage
Solar Array	Requires Deployment and Sun Orientation	Complete Range of Sizes, Easily Paralleled	Must be Integrated with Vehicle Structure and Deployment and Orientation Systems	$600,000 per Kilowatt Ten Year Life, No Additional Cost
Radioisotope Thermoelectric Generator	Requires Shielding Heat Rejection View, and Thermal Isolation from the Vehicle	Comes in 150 Watt Size but Easily Paralleled	Radiation Hazard High Heat Source	$20,000,000 per Kilowatt, Ten Year Life, No Additional Cost Except Handling & Storage
Brayton Cycle	Requires Shielding, Heat Rejection View, and Thermal Isolation from the Vehicle	Comes in 600 Watt Size but Easily Paralleled	Radiation Hazard Heat Source, Dynamic System	Not Developed $4,000,000 per Kilowatt, Handling & Storage Costs
Shuttle Power	May be Rate and Energy Limited, Subjected to Outages, Requires Electrical Interface	Dependent on Shuttle Availability	Two Terminal Interface	Dependent on Shuttle Availability

Table 4.1-3. Battery Options

Primary	Wh/Lb	Remarks
AgO - Zn	50 - 100	Widely Used
Thermal	5 - 7	Widely Used
Li SO_2	180 - 200	Proposed for Interplanetary Flight
Secondary		
Ni - Cd	10.6 - 17.7	Used for All Long Life Programs
AgO - Zn	30 - 60	Used for Short Missions (Non Magnetic)
AgO - Cd	20 - 30	No Longer Being Proposed
Ni - H_2	13.5 - 22	Being Developed
Ag - H_2	21 - 35	Being Developed

4.1.2.2 Electrical Power Subsystem Design Summary

The primary electrical power for the STR and payload equipment will be provided by eight packs of Lithium Sulfur Dioxide ($LiSO_2$) batteries which can be utilized as appropriate for each mission. The maximum module panel capability provided by this system will be 37.8 kWh with a peak load capability of 3.0 kW for 45 seconds. Bus voltage will be maintained at 28 ± 4V dc. The total energy provided by this battery power supply is slightly lower than the worst case power requirement. However, modular design is being used for the power supply so that battery packs may be added or deleted depending on the power requirements for the experiment being considered. A simplified block diagram of the Electrical Power Subsystem is shown in Figure 4.1-1.

Power distribution and control will be via a central power controller and distribution panel which provides the interface between the primary batteries and the subsystems. The ground power connections will also be made through this equipment. All power switching will be accomplished by commands from the command decoder. STS crew override capability is provided to disable any power controller functions in the event of malfunction or emergency.

All payload equipment requiring other than 28±4 volts power must provide self-contained power converters and regulators. These secondary power systems will be electrically isolated from the primary bus. Both primary and secondary systems will be isolated from vehicle structure except at the vehicle unipoint ground.

4.1.3 ELECTRICAL POWER SUBSYSTEM COMPONENTS

4.1.3.1 Lithium Batteries (Lithium Sulfur Dioxide)

Lithium batteries ($LiSO_2$) were selected for the electrical power system because they offer:

1. Higher Energy Density
2. Superior cold temperature performance

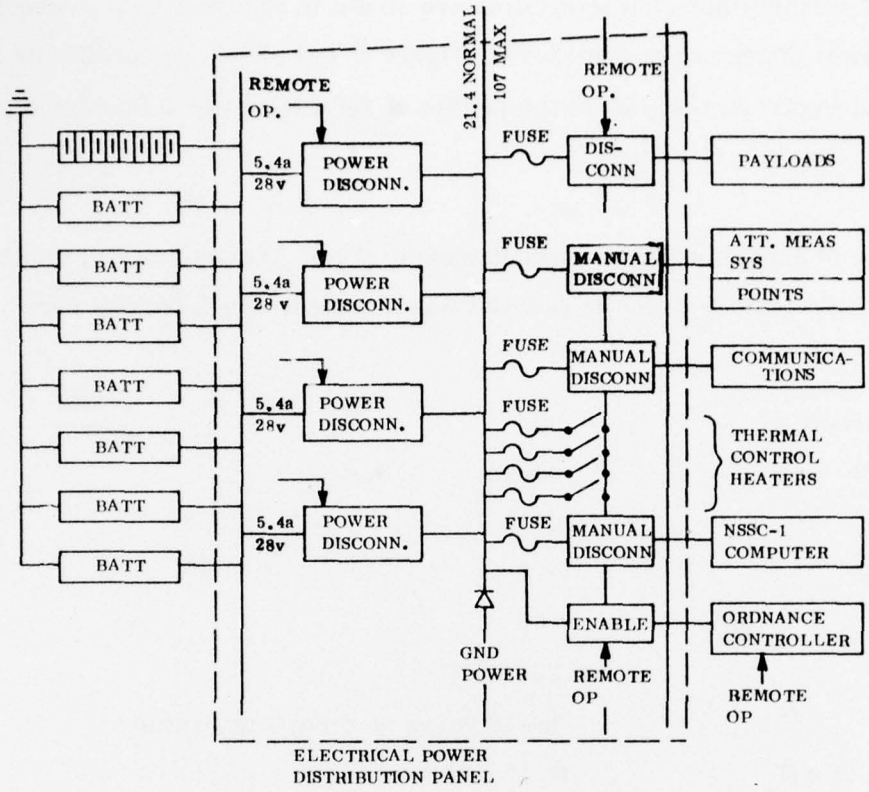


Figure 4.1-1. Block Diagram of Electrical Power S/S

3. Long Active Shelf Life

4. Cost Effectivity

Lithium, having the highest potential of metals in the electromotive series, provides an operating voltage about twice that of traditional systems. The $LiSO_2$ battery cell has an open circuit voltage of 2.92 volts. The battery is a hermetically sealed cell. GE RESD is currently working with Honeywell Power Center, Horsham, Pa. in the development of the selected 160 AH cell which is being-considered for use on the Navy RDSS program. Battery characteristics are as follows:

$LiSO_2$ characteristics

cell size	160 AH to 180 AH
watt hour/lb	150
watt hour/in^3	7.7
dimensions	6.3" high x 3.375" dia.
weight	3.0 lb

Typically $LiSO_2$ cell performance characteristics are shown in Figure 4.1-2 (Percent of rated capacity versus discharge temperature), Figure 4.1-3 (discharge profile as a function of load), and Figure 4.1-4 (Discharge profile of 160 AH cell as a function of load.)

To meet the majority of experiment power requirements (10 of 12 experiments) for the STR, a power supply of 8 battery packs in parallel was selected. Each battery pack is rated as follows:

No. of cells (series)	12
Voltage Range dc	24-32
Amp Range dc	0-14
Normal Amps dc	2.7
Amp Hours	160
Temp. Range	30-125°F
Shelf Life	10-12 years at room temperature
Dimensions L x W x H	9.5" x 18" x 7"

Therefore using 8 battery packs provides the following power and energy:

Voltage	28 Vdc Nominal
Amps	21.4
Watts	600 Nominal
AH	1440
kWhr	37.8
Approx. dimensions	38 x 32 x 7 inches
Approx. weight	288 lbs.

Modular design is provided so that battery packs may be added or deleted depending on the power requirements for each Payload.

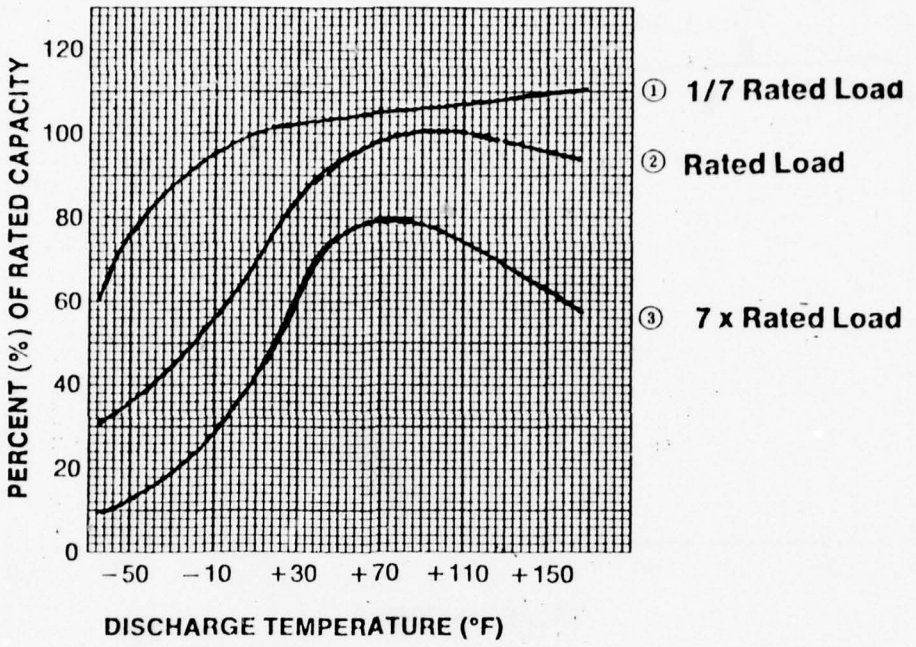


Figure 4.1-2. Typical $LiSO_2$ Cell Performance Characteristics

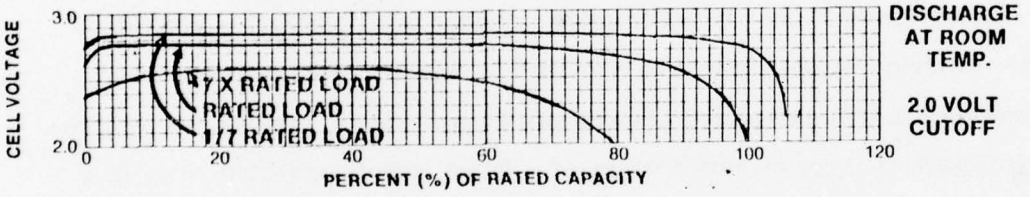


Figure 4.1-3. Discharge Profile as a Function of Load

4.1.3.2 Electrical Power Controller and Distribution Panel

The electrical circuit diagram for the Electrical Power controller and Distribution Panel is shown in Figure 4.1-5. The power control and power distribution for the STR will be provided through this panel. By using modular construction of four pairs of battery packs, the power system can be sized as appropriate for each mission. Four remote controlled 35 amp disconnect switches similar to switches used on the Shuttle Experiment Power Switching Panel (EPSP) will connect the batteries to the main power bus. The battery disconnect switches are dc circuit breakers with magnetic trips. The circuit breakers may be selectively switched for isolating a battery

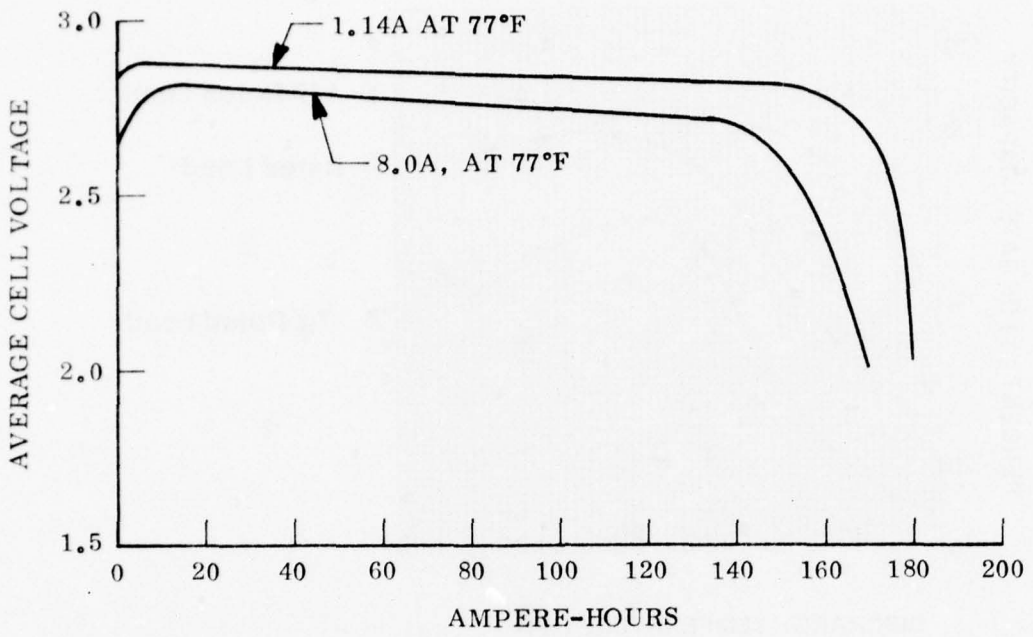
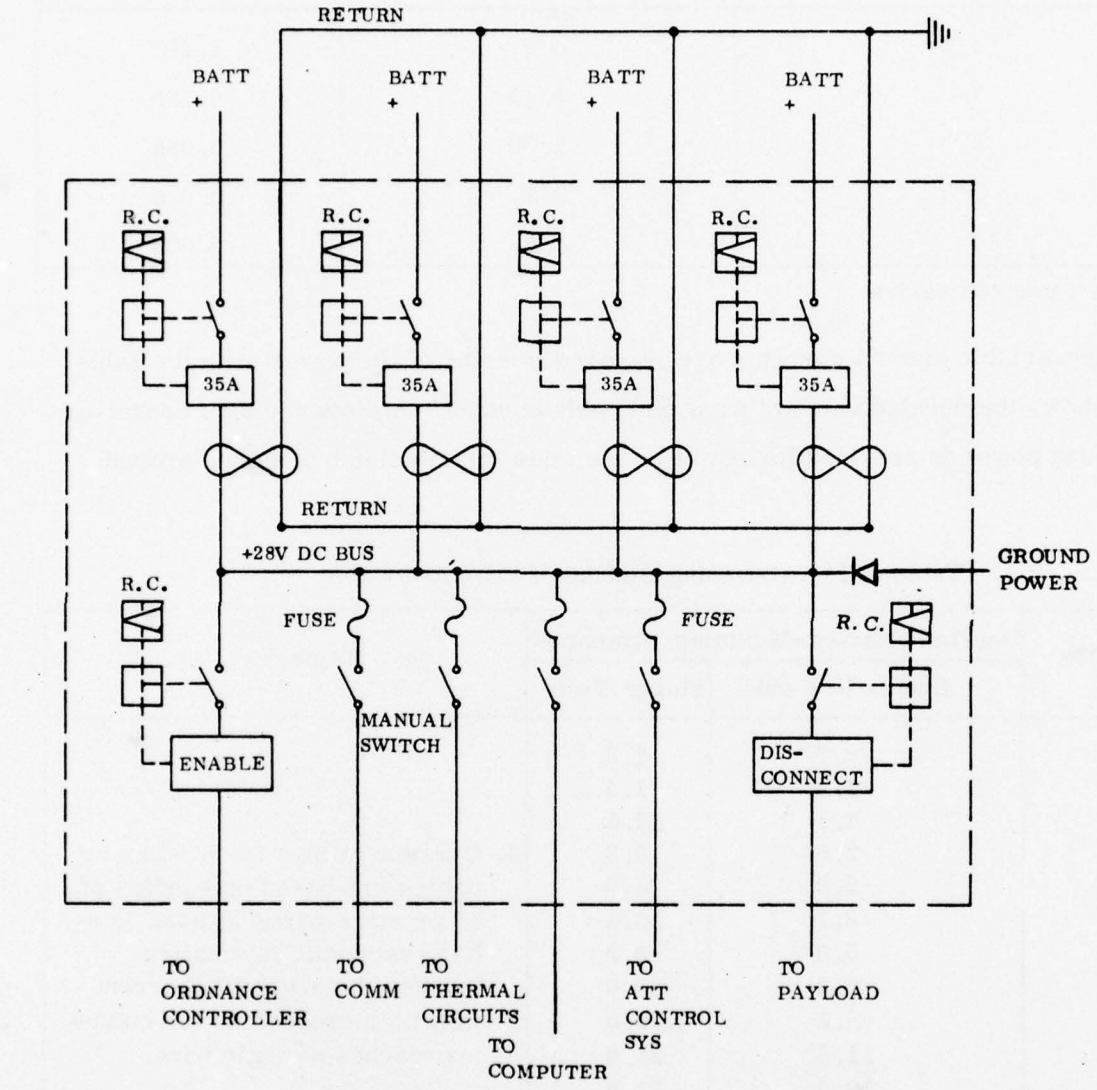


Figure 4.1-4. Typical $LiSO_2$ Cell Performance Characteristics 160 AH Cell

pack or all circuit breakers may be switched simultaneously by on/off commands. The STS crew override capability will be provided to disable these switches in the event of a malfunction or emergency. Power will be provided to the payload by a remote operated magnetic type circuit breaker or a fused remote operated relay (not shown).

For safety requirements, a remote operated magnetic type circuit breaker is required to enable the Ordnance Controller circuit. Two more commands in the proper sequence in the Ordnance Controller are required to fire the electro explosive devices (EED's). Manual disconnect switches with fuses in each circuit provide supply power to all other subsystems. The manual disconnect switches will all be closed prior to launch. The Thermal Control Subsystem has thermostats to automatically control the temperature of each subsystem. Table 4.1-4 lists the fuse rating with a typical circuit rating. The power source will be capable of supplying at least twice the current rating of the fuse. Fuse selection will consider also turn-on/off current transients.



NOTE

R.C. — REMOTE CONTROL CIRCUIT BKR.
RETURNS NOT SHOWN ON LOADS

Figure 4.1-5. Electrical Power Controller and Distribution Panel

Table 4.1-4. Fuse and Circuit Rating

Current Rating of "PICO FUSE"* (AMPS)	Circuit Steady State Rating (AMPS)	Average Resistance at Rated Current (OHMS)
1/2	0.2	1.20
1	0.45	0.185
2	1.00	0.085
5	2.5	0.020
10	5.0	0.009

* Little Fuse 265 series

Since appreciable size dc currents are required in some of the power circuits Table 4.1-5 shows the derated value of wire and cable bundles. A diode isolated connector for ground power is provided for power to the main distribution bus during ground test.

Table 4.1-5. Derating Outline of Wire and Cable

Wire Size	Derated Current-Maximum Amperes		Remarks
	Bundle or Cable	Single Wire	
30	0.7	1.3	1. Current ratings for bundles or cables are based on bundles of 15 or more wires at 343K in a hard vacuum. In smaller bundles the allowable current may be increased as the bundle approaches a single wire. 2. Ratings are based on Teflon insulated wire (Type TFE).
28	1.0	1.8	
26	1.4	2.5	
24	2.0	3.3	
22	2.5	4.5	
20	3.7	6.5	
18	5.0	9.2	
16	6.5	13.0	
14	8.5	19.0	
12	11.5	25.0	
10	16.5	33.0	
8	23.0	44.0	
6	30.0	60.0	
4	40.0	81.0	
2	50.0	108.0	
0	75.0	147.0	
00	87.5	169.0	

4.1.3.3 Harness

The standard test rack harness provides the electrical interfaces within and between the supporting subsystems and the experiments. It is critical element in the performance of the test rack and more precisely in the performance of the experiments.

Figure 4.1-6 shows a typical Nimbus/Landsat harness installation.

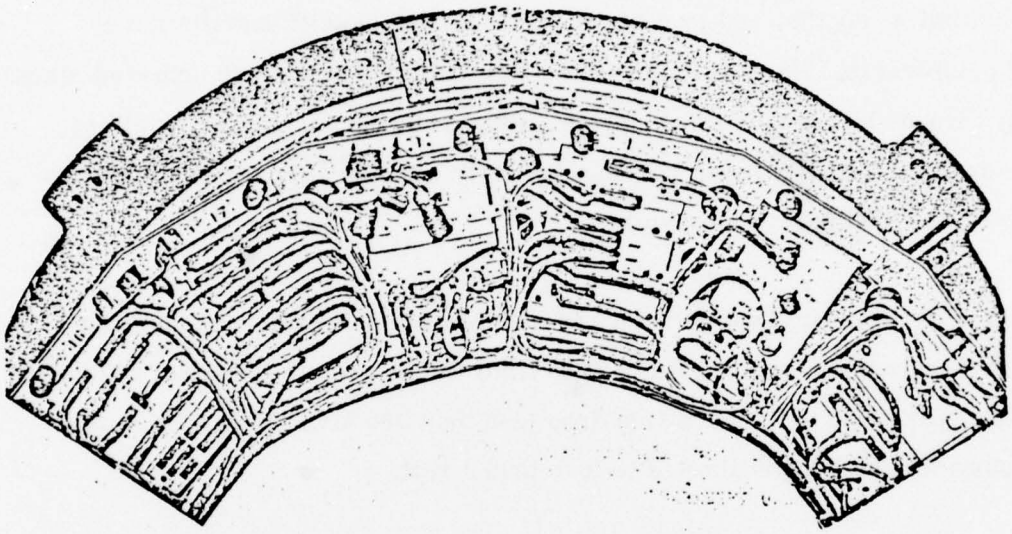


Figure 41.-6. Typical Nimbus/Landsat Harness Installation Section

The harness is divided into support subsystem harness and payload harnesses. A single universal harness is used for all experiments. The harness is physically configured into separate functional signal types:

1. power distribution
2. telemetry
3. command

4. pyrotechnic

5. signal

This provides a maximum isolation of signals. The harness will be designed on a full scale three dimensional mockup to optimally size all wire routings. Flight proven component are used throughout, and selective redundancy is provided for critical functions.

Component locations are selected to be consistent with electrical and thermal requirements, to minimize lengths, and to route without interfering with installation or removal of components. Harness interconnections between panels are achieved simply and reliably through single point disconnects. Requirements for mutual shielding, shield terminations, and routing of harness is easily satisfied. Critical signals are run as twisted shielded pairs or coaxial cables.

Wire gauge for each power line and return is sized to carry the current, operate within an acceptable temperature rise, and to limit the IR drop to less than 280 millivolts. Loads over 100W will have a line drop less than 500 MV. Line drop to the Thermal subsystem heaters is limited to less than 1 volt.

The use of non-magnetic connectors and minimum use of magnetic components in the fabrication and installation of the harness segments and twisting of current carrying wires (electric field cancellation) minimizes the magnetic moment.

A single point grounding scheme is used for power distribution with the ground point position to cause minimum EMI. Transponders are grounded to the structure to establish a solid RF reference. DC/DC converters are used to isolate the signal ground from the power return. Telemetry ground is used as a reference for the low level telemetry housekeeping and monitoring signals only. All components are electrically bonded to the spacecraft structure.

Reliability considerations dictated formulation of the wiring redundancy philosophy for the harness subsystem. The major decisions are:

1. Preliminary power to a component containing a prime and backup unit (redundant component) are supplied by independently routed twisted pairs
2. Power to a prime (only) component is supplied by two twisted pairs
3. Clock and signal lines to redundant components are by independently run wires
4. Mission critical telemetry will be routed separate from independent telemetry drivers

4.1.3.4 Ordnance Controller

The ordnance controller contains relays to switch power from the batteries to the electro-explosive devices (EEDs) and is the same ordnance controller used on BSE (Broadcast Satellite Experimental) - a GE built, Japanese satellite.

The EEDS have a minimum all-fire current of 3.5 amperes for at least 10 milliseconds. A current limiting resistor is selected to assure a minimum of 5 amperes and a maximum of 15 amperes for the full range of battery voltages and bridge wire impedance over the operating temperature range.

In addition to the ordnance controller bus enable in the power controller, two relays are in series with each EED to preclude premature actuation of these functions. The first relay is a latching relay and serves as an enable or arming function, the second is a holding relay that will pull in for the duration of the command pulse firing the EED.

Circuits which are connected to the EEDs prior to actuation are balanced, twisted and shielded to prevent actuation by radiated interference. High value resistors are used to prevent build up of static charge on the lines to the EEDs. For redundancy batteries with redundant cables are connected directly to the ordnance controller. Any open circuit failure between the battery and the EED or open relay coils or inability

to send a command in the primary channel is avoided by using a redundant backup channel. The Ordnance Controller allows verification of proper electrical installation of the EEDs by providing for low current resistance measurement of the EED, harness and connectors through the normally closed contacts of the fire relays. Telemetry verification of the arming and firing functions is provided.

All pyrotechnics and pyrotechnic firing circuits meet the range safety requirements of ETR and the STS as shown in Table 4.1-6.

Table 4.1-6. Pyro Characteristics

Bridge Wire Resistance	1.1 ± 0.1 ohm
Minimum All-Fire Current	3.5 Amp for 10 ms
Maximum No-Fire Current	1 Amp/1W for 5 min

There are two separate cartridges, prime and redundant for every pyro, each with its own bridgewire. Firing or misfiring of one will not affect its backup. This precludes the necessity for simultaneity in firing.

4.1.3.5 EMC Considerations

EMI Shielding. Individual harness segments will be EMI classified and separated to reduce capacitive and inductive coupling.

RF Signals. RF signals will be distributed via waveguide, flexible coax or semi-rigid coax cable. The shield of the coax will be well grounded at both ends for compliance with radiated emission limits.

Digital Signals. Wire handling clock and enable signals for magnitude commands will be twisted pairs with shields grounded at both ends. Wire handling magnitude commands will be single ended with shields grounded at both ends.

Analog Signals. Analog signals will be distributed in twisted pairs with shields grounded at the receiver end of the circuit.

Discrete Level Signals. The command lines from the decoder to various subsystems will be shielded in discrete bundles as required. These wires will be routed away from all sensitive lines. The shields will be grounded at the signal source end.

Power Lines. All power lines will be twisted. Primary dc power lines will not be shielded. Shielding may be utilized on secondary dc power lines, where necessary, to contain noise or to protect sensitive equipment from external noise. Such shielding will be grounded to structure at the load.

Electroexplosive Device (EED) Firing Circuits. Firing circuit conductors will be twisted pair shielded. EED firing circuits will be balanced and isolated from other electrical circuits and from each other by means of individual shields. Shielded EED circuits will be routed together in common secondary shields which will be multipoint grounded. Shields will be peripherally grounded to the EED connector case. No EED circuit will share ground return with other circuits.

Harness Routing. In the design of the harness, the wires of each EMI class will be run in separate bundles. Where possible, each cable classification will be physically separated. Cables will be routed as close to the structure as possible, and will be clamped at frequent intervals to minimize generation of electromagnetic fields due to wire movement relative to the structure. EMI shielding will be used to suppress radiated or received interference and to prevent EMI coupling between noisy and susceptible wires and cables.

4.2 COMMUNICATIONS SUBSYSTEM (CS)

The STR provides telemetry, command, and data handling links to all experiments direct from the ground.

4.2.1 COMMUNICATION REQUIREMENTS

Requirements for all the experiments are summarized as follows.

1. Most Experiments

   a. Telemetry and Command

      - SGLS Carrier I
      - Telemetry on 1.024 MHz Subcarrier Oscillator (SCO)
      - 2 watt Transmitter
      - Two element turnstile antenna; 6 dBi gain

   b. Payload Data

      - SGLS Carrier II
      - Data Rate of 1.024 Mbps
      - 2 watt Transmitter
      - Turnstile antenna; -3 dBi gain

2. Lasercom

   a. Telemetry and Command

      - SGLS Carrier I
      - Telemetry on 1.024 MHz SCO
      - 2 Watts Transmitter
      - Two Element Turnstile Antenna; -6 dBi gain

   b. Payload Data

      - SGLS Carrier II
      - Data Rate of 1.024 Mbps
      - 2 Watt Transmitter
      - 1.9' diameter antenna (20 dB gain) Mounted on 2 Axis Gimbal Assembly with Open Loop Pointing

4.2.2 COMMUNICATION SUBSYSTEM (CS) DESIGN SUMMARY

Since the requirements of the STR and its experiments fall easily within the capabilities of existing Space Ground Link System (SGLS) equipment, the CS has been designed to take maximum advantage of this hardware.

The CS consists of the equipment shown in Figure 4.2-1 which provides for the transmission and reception of data between the STR and AFSCF. All communications will be SGLS compatible at S-Band and the functions of the CS will be to provide:

1. Transmission of payload data to the AFSCF
2. Transmission of return ranging and telemetry signals to the AFSCF
3. Reception of ranging and commands signals from the AFSCF
4. Transmission of beacon signal to the AFSCF
5. Storage of telemetry and payload data

A description of the necessary hardware changes to provide a STDN compatible communication subsystem is shown in Appendix A.

Communications equipment with a system margin of 3 dB will be used for AFSCF elevation angles greater than 5 degrees with a BER performance of 10^{-5} or less on the payload data, telemetry and command links. The telemetry unit will multiplex telemetry from the STR subsystem and experiments baseline configuration, the telemetry data is transmitted as plain text. Provisions will be made to add an encrypter such that the telemetry data may be transmitted as cyphered text. When the STR is not in contact with a tracking station, telemetry will be recorded on a tape recorder and will be played back during station contacts via the SGLS carrier II link. Many of the critical CS components are redundant as shown in Tables 4.2-2 and 4.2-4.

Uplink signals will be received via turnstile antennas mounted on opposite sides of the STR to provide near-spherical coverage and redundant receivers will provide carrier lock and demodulate command signal capabilities. The uplink cyphered command for

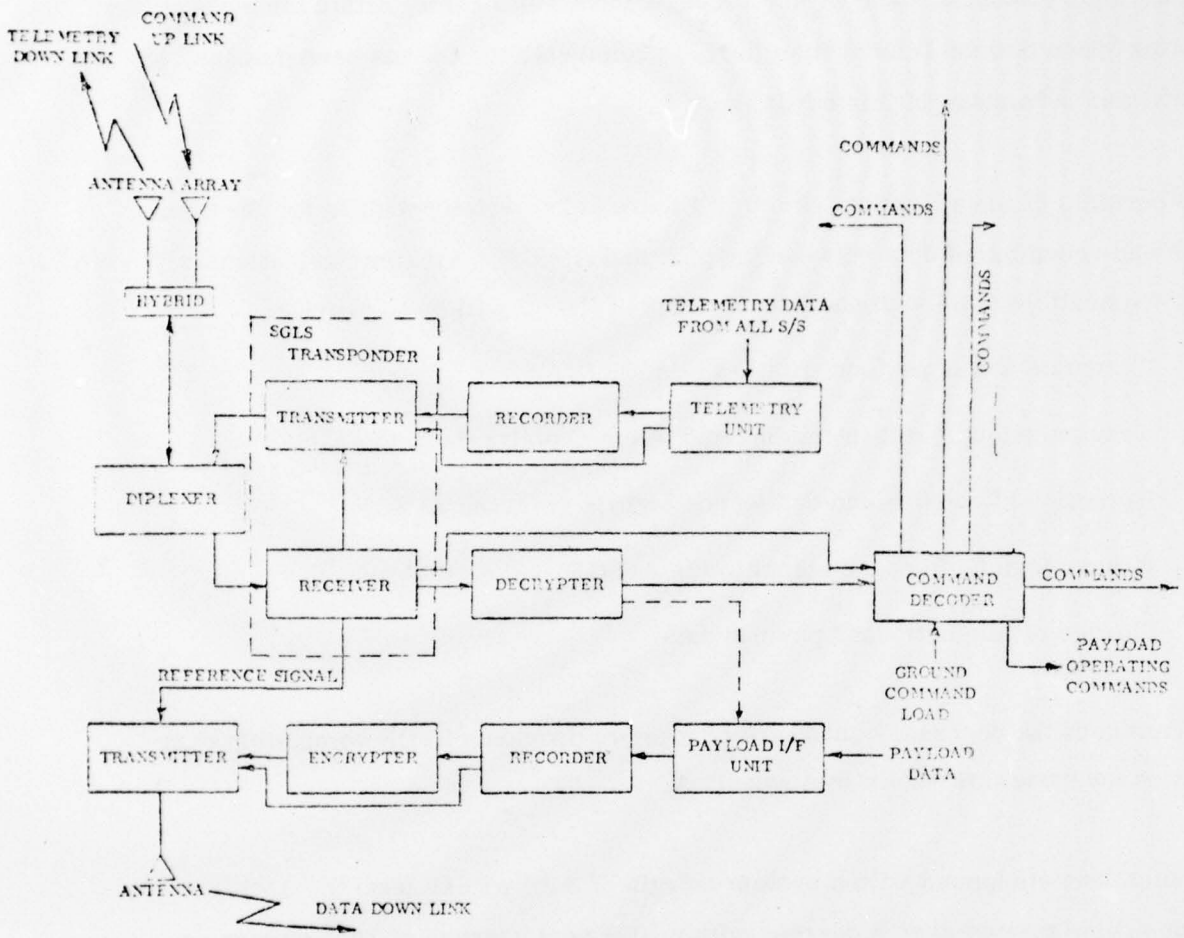


Figure 4.2-1. Communication S/S Block Diagram

STR housekeeping and experiment control will be decrypted and fed to the Command Decoder where decoded commands are distributed to experiment users.

The command decoder and telemetry unit to be used within the CS has the following characteristics:

1. Command Decoder

   a. Discrete Commands - 1024

   b. Serial Commands - 48

- Magnitude Commands of 14 Bits
- Message Commands of 18 Bits

2. Telemetry Unit

a. Inputs

- Discrete (Bi-Level) 350
- Serial 18
  8 bit magnitude
- Analog 250
  8 bit resolution

b. Output

• Main Frame Matrix	256 columns x 30 rows
• Main Frame Rate	1 main frame/61.44 seconds
• Max. Word Sampling	256 Samples/minor frame
• Bits Per Word	8
• Bit Rate	1000 bps
• PCM Encoding	NRZ-L
• Minor Frame Rate	1 minor frame/2.048 seconds

Link calculations were made for the downlink service of telemetry and experiment data for most missions and the Lasercomm mission. The BER performance requirement of $<10^{-5}$ with a 3 dB system margin is achieved for all links. A high value of ground ERP is available which will provide sufficient margin to make all uplinks non-critical.

The link calculations were made for communications with the SCF 60-foot antenna rather than the 46-foot antenna. The G/T value for the 60-foot antenna is less than the 46-foot antenna G/T value and therefore represents a worse case condition. Operation with the Thule SCF was not considered except for the station contact time analysis.

The telemetry and command requirements are met with a 2-watt transmitter and -6 dBi antenna operating with any of the 46-foot or 60-foot antennas of the AFSCF. A summary of the equipment characteristics are given in Table 4.2-1. Size, weight, and power requirements are listed in Table 4.2-2.

Table 4.2-1. Telemetry and Command Equipment Characteristics

- Operate with AFSCF; SGLS compatible
- Command at 1 Kbps; Store in on-board computer
- Telemetry
  - Real Time Direct
  - Play Back Tape Recorder
- Two-element antenna array; near-spherical coverage
- Redundant Channels
- Total Weight 64 pounds
- Total Power Required 46 Watts

4.2.3 COMMUNICATION SUBSYSTEM COMPONENTS

4.2.3.1 Antenna

The STR antennas will consist of two turnstile elements mounted on opposite sides of the STR and connected by a hybrid. Based on tests of these elements on other vehicles, it is anticipated that coverage to the -6 dBi level will be obtained over 80% of the total STR volume. A single turnstile antenna will be employed for all experiments except Lasercom, that will provide a coverage area to the -3 dBi signal level of approximately ±90° from boresight. Figure 4.2-2 shows the turnstile antenna which was designed for both the Japanese Broadcast Satellite and DSCS III Programs.

RF transmissions in the shuttle bay is a concern. Further investigation is necessary and should include options of placing the antennas on deployable booms to minimize the shuttle interfaces.

4.2.3.2 Receiver

The receiver will operate on an assigned uplink frequency in the band of 1760 to 1840 MHz and will have a noise figure not in excess of 7.5 dB. The noise figure includes a 3 dB loss due to a hybrid used for power dividing the input signal between the redundant receivers. The receiver demodulates the carrier and applies the command tones to a detector. The tones are detected and command data and sync signals are provided to the command decoder. Upon carrier lock and AM sync detection, the receiver provides a turn-on signal to the command decoder and other STR equipment as required.

Table 4.2-2. Typical Telemetry and Command Hardware Matrix

Item	Qty	Size (Each) In.	Weight (Each) Lbs.	PWR (Each) Watts Operate/Standby	Typical Equipment Heritage
Antenna	2	5 Dia x 5	0.2	--	GE Design - Japanese B SE, DSCS III
Hybrid (ANT)	1	1.3 x 1.5 x 0.4	0.04	--	Anaren 10040-4, DSCS III
Diplexer	1	2.0 x 2.4 x 7.8	1.5	--	WAVECOM S-204, DSCS III
Switch	1	2.8 x 2.3 x 1	0.38	--	TRANSCO 909C72400, DSCS III
Transmitter	2	6 x 7.2 x 1.7	3.0	13.5	Cubic TR-37, DSCS III
Telemetry Unit (Redundant)	1	8.5 x 11.6 x 6.0	13.4	8.0	GD, DSCS III
Recorder	2	3 x 6 x 7.8 (Tape Trans.) 4.5 x 6 x 7.4 (Electronics)	10.0	10.0	RCA, TIROS-N, NIMBUS-G, SEASAT
Hybrid (RCVR.)	1	1.3 x 1.5 x 0.4	0.04	--	Anaren 10040-3, DSCS III
Receiver	2	6 x 7.2 x 1.7	3.0	2.0	Cubic TR-37, DSCS III
Decrypter	2	5.8 x 3.6 x 6.0	4.4	2.2	GFE-KIR - 23A
Command Decoder (Redundant)	1	8.5 x 6.3 x 6.0	7.6	10.0/4.0	GD, DSCS III

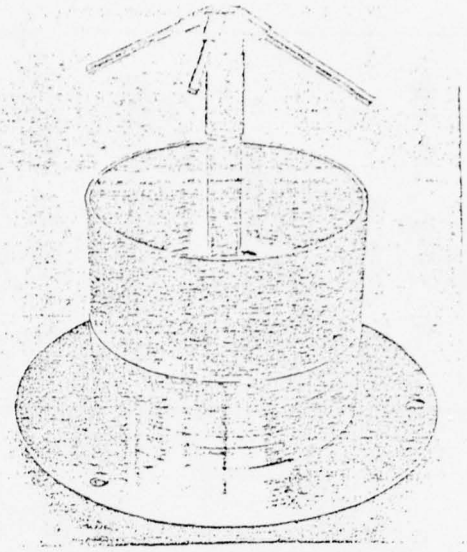


Figure 4.2-2. GE Design S-Band Omni Antenna

4.2.3.3 Command Decoder

The command decoder processes, validates, and distributes discrete and serial commands to the various STR subsystems. All functions of the command decoder are block redundant and command processing paths are determined by a configuration command to the command decoder. The command decoder operates in a low power mode when not receiving input signals. Command verification is accomplished by telemetering the message portion of the command data received by the processor via words in the main telemetry frame. In addition, the command section provides a decoder verification output to the downlink telemetry. An "accept" will be sent if the command pulse is issued, and a "reject" if processing was initiated, but failed to result in a valid command pulse. Commands are executed in real time or stored with execution at a later time when the STR is not in contact with a tracking station.

4.2.3.4 Telemetry Unit

The primary function of the telemetry unit is to sample and encode analog, digital, and discrete STR diagnostic and experiment operational data into a pulse code modulation (PCM) time multiplexed format suitable for modulation of the S-band subcarrier. It also determines the proper power turn-off times of the encryption and decryption equipments. All telemetry timing, clock and synchronization signals are provided by the telemetry unit.

4.2.3.5 Transmitter

The maximum data rate of the experiments examined is defined as 512 Kbps for 10 minute data gathering periods. A tape recorder will be required to store the data until satellite contact with an AFSCF ground station can be accomplished. It is estimated that an average contact time of 7.3 minutes exists between the satellite and the ground station. In order to read out the payload data during the contact time period it is necessary to playback the data at a higher rate than it is recorded. A playback data rate of 1.024 Mbps is selected for compatibility with station contact time duration, STR tape recorder and AFSCF demodulation capabilities. The transmission of the data at 1.024 Mbps requires that the SGLS Carrier II be employed and this requires the need for the experiment transmitter channel equipment depicted in Figure 4.2-1. If the experiment data were to be restricted to data rates below 256 Kbps then the information could be transmitted to the AFSCF via the telemetry transmitter by utilization of the 1.7 MHz subcarrier thereby eliminating the need for a experiment transmitter.

Figure 4.2-3 illustrates the maximum data rate versus satellite altitude that the AF SCF can support for a satellite using a 2 watt transmitter and a omni (-6 dBi) antenna. The 60' and 46' AFSCF stations are compatible with data rates of 1.024 Mbps for satellite altitudes up to 800 nm.

The above discussions concerning data rates and associated hardware were based upon the assumption that the experiment must be processed within one orbit period by any one of the AFSCF stations. If the requirements allow several orbital periods for complete processing of experiment data then the data rate may be reduced. A reduction in the data rate improves the system performance in that the SGLS Carrier II service is not required, thereby eliminating the experiment data handling equipment. (The experiment data would be transmitted via the 1.024 or 1.7 MHz subcarrier).

The experiment data transmission requirements, can be met with a spacecraft 2 watt transmitter and -3 dBi antenna operating with any of the 46' or 60' antennas of the AFSCF.

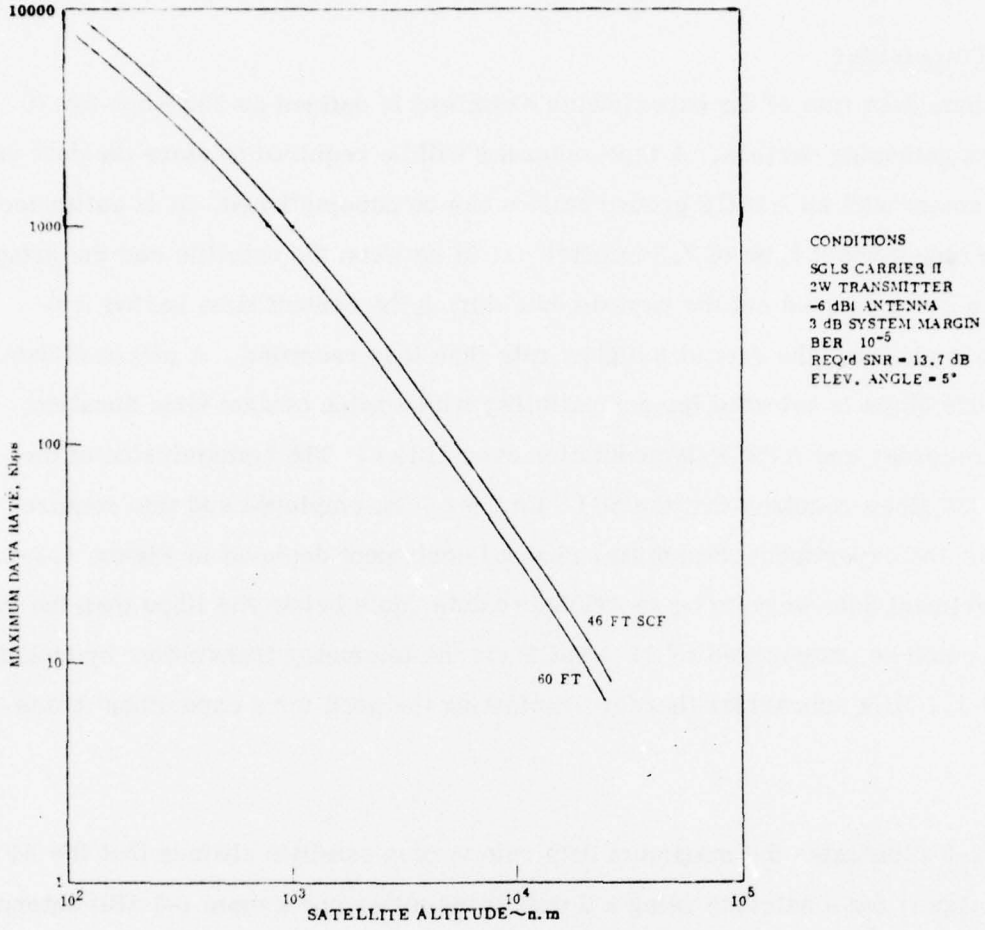


Figure 4.2-3. Data Rate vs. Altitude

A summary of the experiment data equipment characteristics are given in Table 4.2-3. Size, weight and power requirements are listed in Table 4.2-4.

Table 4.2-3. Experiment Data Equipment Characteristics

- Operate with AFSCF; SGLS Compatible
- Data Rates Up to 1.024 Mbps
- 2 Watt Transmitter
- Antenna
  - All experiments except Lasercom use a turnstile design resulting in -3 dBi gain over ± 90° from boresight
  - The Lasercom experiment uses a steerable dish antenna (1.9' dia. max.) resulting in a 20 dB gain
- Record Capability
- Encryption Capability
- Redundant Channels
- Total Weight 35 pounds
- Total Power Required 30.5 watts

Table 4.2-4. Typical Experiment Data Hardware Matrix

Item	Quantity	Size (Each) In.	Weight (Each) Lbs.	Power (Each) Watts	Typical Equipment Heritage
Antenna	1	5 Dia x 5	0.2	---	GE Turnstile, B SE and DSCS III
Switch	1	2.8 x 2.3 x 1	0.38	---	TRANSCO 909 C72400, DSCS III
Transmitter	2	6 x 7.2 x 1.7	3.0	13.5	Cubic TR-37, DSCS III
Encryptor	2	5.4 x 4.3 x 5.8	4.3	7.0	GFE - KG 28 A
Recorder	2	3 x 6 x 7.8 (Transport) 4.5 x 6 x 7.4 (Electronics)	10.0	10.0	RCA, TIROS-N, NIMBUS-G, SEA SAT

The transmitter selected to handle the data output is two watts at an assigned carrier frequency in the band of 2,200 to 2,300 MHz. Two subcarriers (1.024 and 1.7 MHz) are available for biphase modulation by the PCM telemetry signal. The source of the data will be either real time from the telemetry unit or play back from the tape recorder. Only one transmitter will operate at a time. A latching type coaxial switch with inherent low loss and high reliability characteristics is utilized to select the primary or redundant transmitter.

The proposed receiver/transmitter combination (transponder) also has the capability to process (turnaround) the AFSCF uplink PRN ranging code to provide a method for obtaining range or range rate data. The received PRN code at 1 Mbps is combined with either or both telemetry subcarriers and the resultant signal is used to phase modulate the output carrier.

The experiment data will be transmitted via the SGLS Carrier II Service which is 5 Mhz below the Carrier I Service used for telemetry transmission. A reference signal derived from the SGLS receiver is used to establish the transmitted frequency. The transmitter generates the output frequency (f_T) as follows:

$$f_T = \frac{256}{205} f_o - 5 \text{ MHz}$$

where

f_o = receiver input frequency.

The AFSCF Carrier II Service is capable of supporting the transmission of data for rates up to 1.024 Mbps and is essentially the same design described previously for the telemetry/ranging transmitter except that the telemetry input/modulator circuits are not required. The experiments data would be handled as in the PRN code where the data directly modulates the carrier II frequency. The modulation index is nominally $\pi/2$ radians in order to achieve optimum performance.

4.2.3.6 Tape Recorder

Predicated upon the various experiment data characteristic the tape recorder will be programmed to accept PCM telemetry from the telemetry unit at appropriate times when the STR is not in contact with a tracking station. Recording times will be controlled by stored commands from the onboard computer. During tracking passes, upon direct or stored command, the tape is played back for transmission of stored telemetry data to the AFSCF. The tape recorder capabilities will be as follows:

1. Input data rate (R_1) = 512 Kbps
2. Output data rate (R_o) = 1.024 Mbps
3. Storage Capacity (C) = 3.07×10^8 bits Experiment — $C = R_1T$ where T = record time (600 sec.)

4.2.3.7 Electronic Counter Measures (ECM) Considerations

ECM threats potentially can deny the STR mission if proper measures are not taken.

Three types of ECM threats to be studied are:

1. SIGINT. Determination of signal characteristics to extract the mission data.
2. Spoofing. Capture of command system or entering errors into either command and data links.
3. Jamming. Prevention of mission data to the user by denying communication between satellite and AFSCF's.

As discussed in paragraph 4.2.2 decryption/encryption devices will be utilized on uplink experiment data services. Encrytpion provides substantial protection to the SIGINT and spoofing threats. The baseline does not include encryption of normal STR telemetry data.

A high-signal-to-jammer ratio is used to combat the jamming threat. Increasing the satellite and AFSCF ERP is effective up to a point depending upon the jammer characteristics. This technique is available for the uplink, for example, where nominally the AFSCF command transmitter is operating at a few hundred watts. During a jamming environment, the uplink transmitter power may be increased to 10 kW. However,

this increased ERP may still not be sufficient to combat the jammer and additional protection will be required.

Some protection against the ECM threats are automatically achieved via the use of multiground stations AFSCF's. Additional protection from the ECM threats can also be obtained by the employment of a directive STR antenna.

4.2.4 SGLS GROUND CONTACT

A study was made of the AFSCF ground station contact times for the SGLS. Three STS flight cases were taken which are typical of STS 7-day missions for Spacing in the 1980 to 1982 time period. The orbits are as follows:

	Inclination	Altitude, Nautical Miles	Launch Site
Case 1	28.50	385	ETR
Case 2	57°	325	ETR
Case 3	96°	400	WTR

The ground tracks for the first day are shown in Figures 4.2-4, 4.2-5 and 4.2-6. The contacts and contact time summaries are shown in Tables 4.2-5 and 4.2-6. The smallest contact time occurs at the 57° inclination flight mission where contact occurs on 84% of the 104 orbits. However, even for this case the number of contacts per orbit (19 per day) and the contact times of 3 to 8.5 minutes appear adequate for STR mission requirements. The command rate 1024 bps and the potential stored data dump of 10^9 bits at 1 Mbps using AFSCF Channel 2 appears to be well within the ground system and the STR System capability. Adding the Indian Ocean site to the list of ground stations increases the contacts and times. The Indian Ocean station would make possible shorter response times for STR commands after data receipt and evaluation at other stations. However, it is not a necessary element for the STR mission.

The AFSCF stations are as listed below:

Dual Tracking Stations	Latitude (N)	Longitude (E)	Altitude
NHS - New Hampshire (Manchester, N. H.)	42° 56.9'	288° 22.4'	211 meters
VTS - Vandenberg (Lompoc, California)	34° 49.4'	239° 29.9'	305 meters
HTS - Hawaii (Kaena Point, Oahu)	21° 34.3'	201° 44.1'	287 meters
Single Tracking Stations			
IOS - Indian Ocean* (Mahe, Seychelles)	-4° 40.0'	55° 28.8'	590 meters
GTS - Guam	13° 36.8'	144° 51.2'	161 meters
TTS - Thule (Thule, Greenland)	76° 36.5'	291° 24.0'	141 meters

*Not included in this analysis.

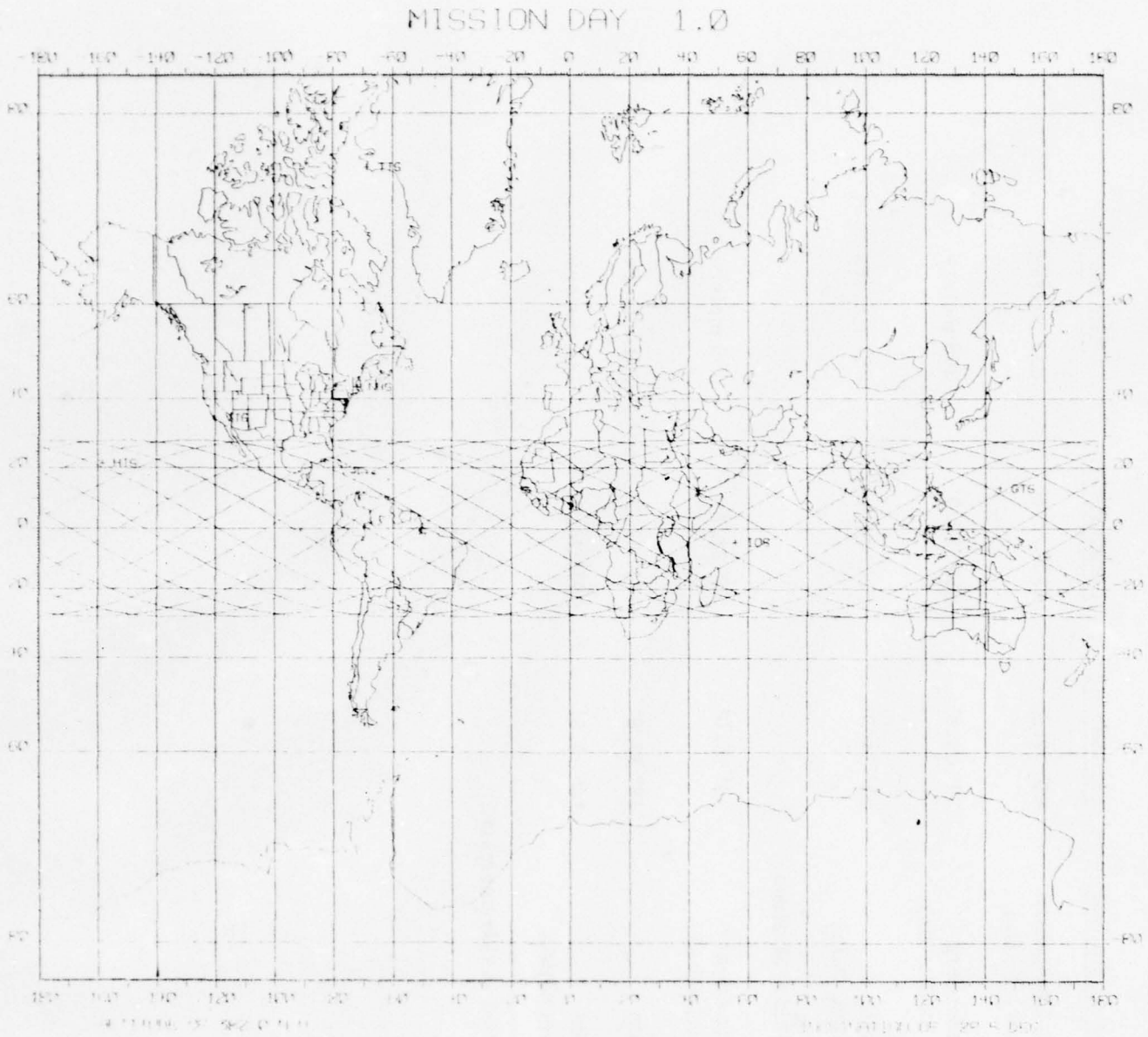


Figure 4.2-4. Ground Track - 28.5° Orbit

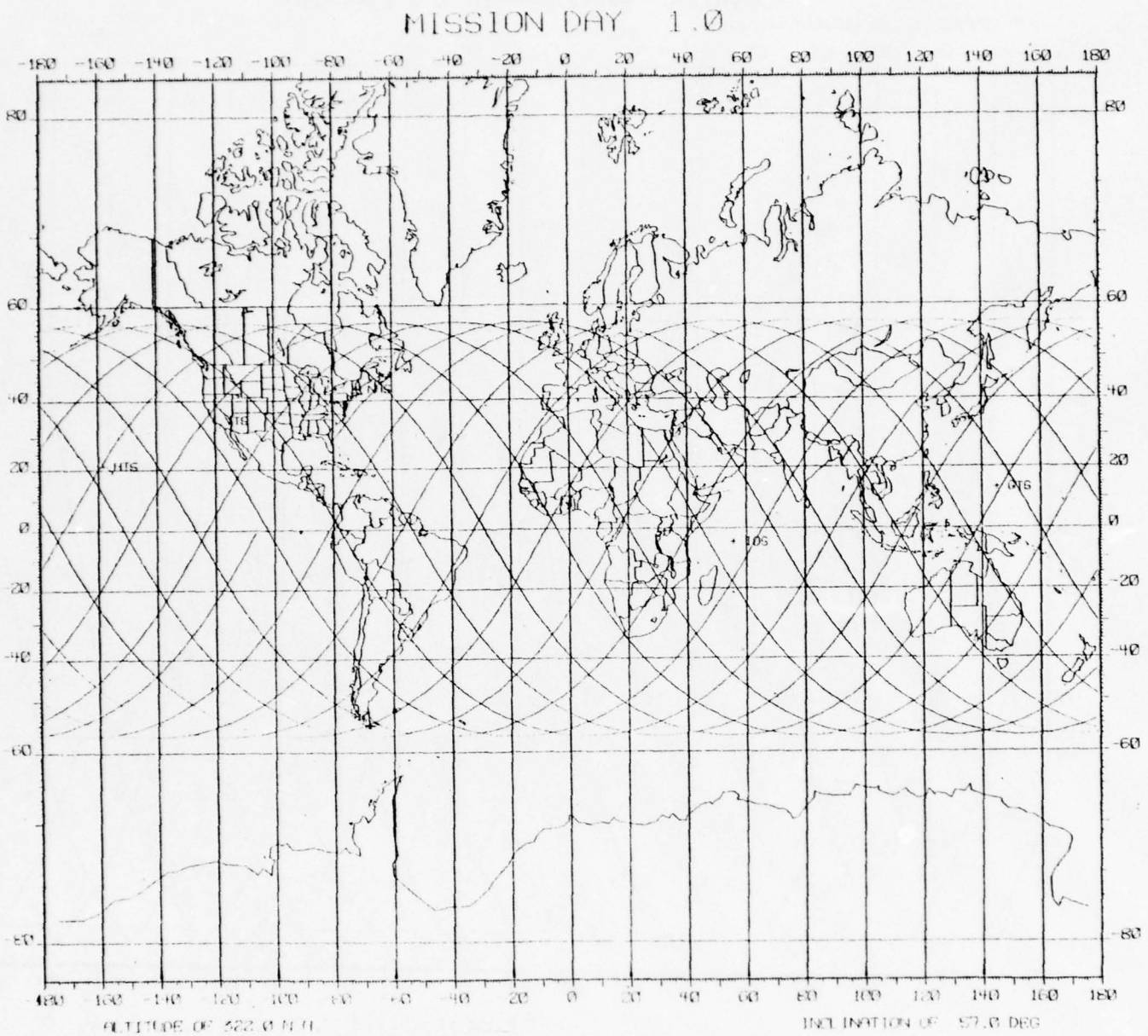


Figure 4.2-5. Ground Track - 57° Orbit

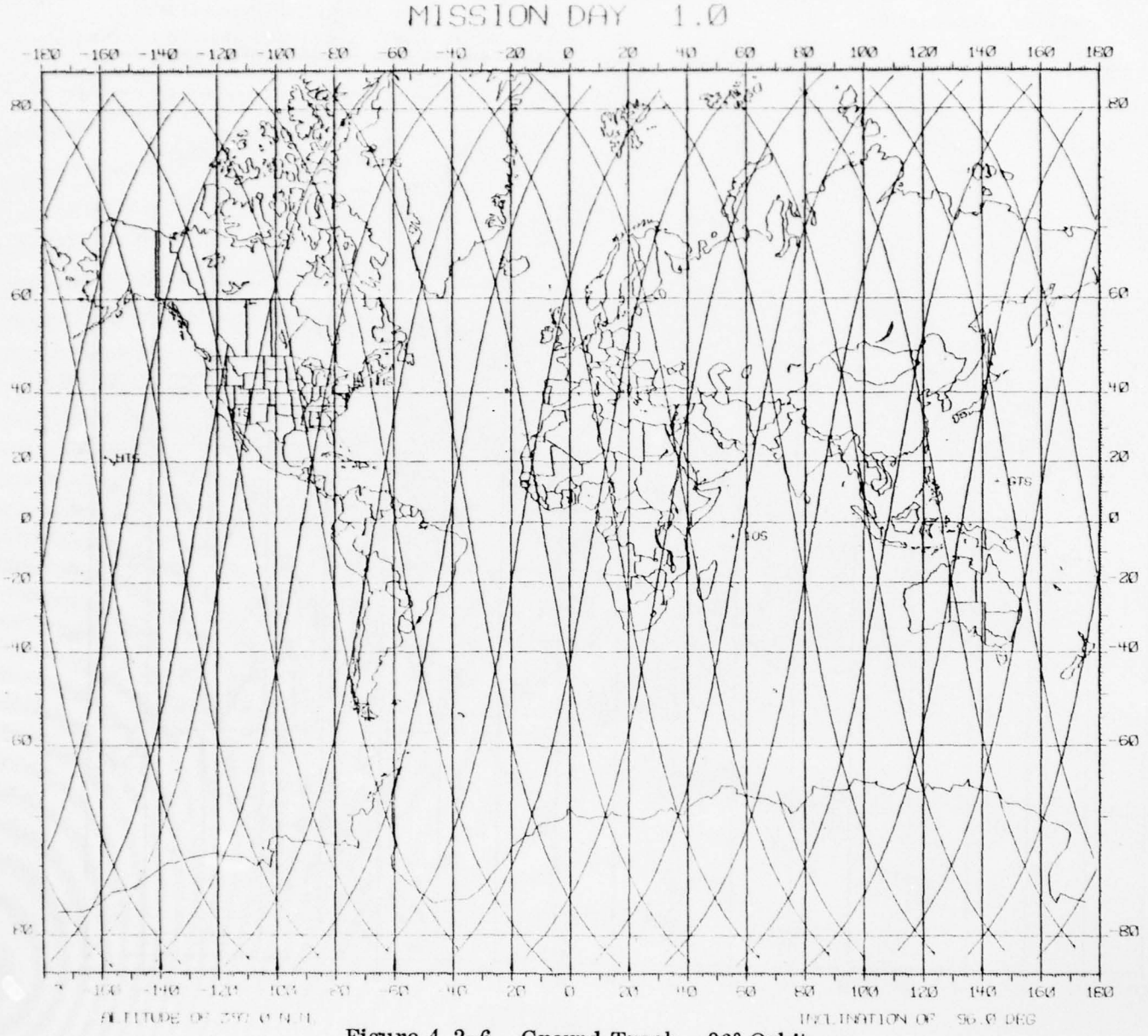


Figure 4.2-6. Ground Track - 96° Orbit

Table 4.2-5. SGLS Ground Contact Summary For 7-Day Flight Without IOS

Station		Total Contact Time, Minutes	Total Number Of Contacts	Contacts Per Day	Number of Orbits with Contact	Typical Contact Time Per Pass Minutes
NHS	28.5°	189	27	3	27	4 to 9
	57°	276	46	5 to 7	46	3 to 8.5
	96°	408	38	4 to 6	38	7.5 to 13.5
VTS	25.5°	355	39	4	39	6 to 12
	57°	183	29	4	29	5 to 8
	96°	361	35	4 to 6	35	5 to 13.5
HTS	28.5°	513	54	7 to 9	54	7 to 12.5
	57°	141	22	3 to 5	22	3 to 8
	96°	314	29	3 to 5	29	8 to 13
GTS	28.5°	586	61	8 to 9	61	3 to 12
	57°	133	20	2 to 4	20	3 to 8.5
	96°	295	28	4 to 5	28	4.5 to 12.5
TTS	28.5°	0	0	0	0	0
	57°	0	0	0	0	0
	96°	1184	100	14	100	5.5 to 11
Total	28.5°	1643	181	26*	84	-
	57°	733	117	17*	86	-
	96°	2562	230	33*	101	-

*Average Value

Table 4.2-6. SGLS Ground Contact Summary for 7-Day Flight With IOS

Station		Total Contact Time, Minutes	Total Number of Contacts	Contacts Per Day	Number of Orbits with Contact	Typical Contact Time Per Pass Minutes
IOS	28.5°	497	57	7 to 9	57	4.5 to 12
	57°	132	20	2 to 4	20	1.5 to 7
	96°	293	28	4 to 5	28	3.5 to 12.5
Total	28.5	2140	238	34*	95	-
	57°	865	137	20*	87	-
	96°	2855	258	37*	101	-

* Average values.

4.3 THERMAL CONTROL SUBSYSTEM (TCS)

The STR provides thermal control services to support subsystems and the payloads.

4.3.1 THERMAL CONTROL REQUIREMENTS

The thermal control design approach for the STR evolved from a systematic study of heat distribution and heat rejection concepts which considered experiment power management and thermal control with the associated STR thermal requirements (maintain 0-30° C with dissipations of 0-1 kw (3 kw peak for 45 sec.) The design was based on maintaining the temperature range of 0°C to 30°C for the STR, Experiment Pointing Module (EPM) and all experiments (as shown in Figure 4.3-1) except the XUV and Teal Ruby. Thermal requirements for independent thermal control of each experiment were determined based on individual experiment evaluations. The power dissipation duty cycle and equipment definition for each experiment for a 7-day mission are shown in Tables 4.3-1 and 4.3-2. Cryogenics are deemed to be experiment peculiar and therefore to be supplied as part of the experiment system.

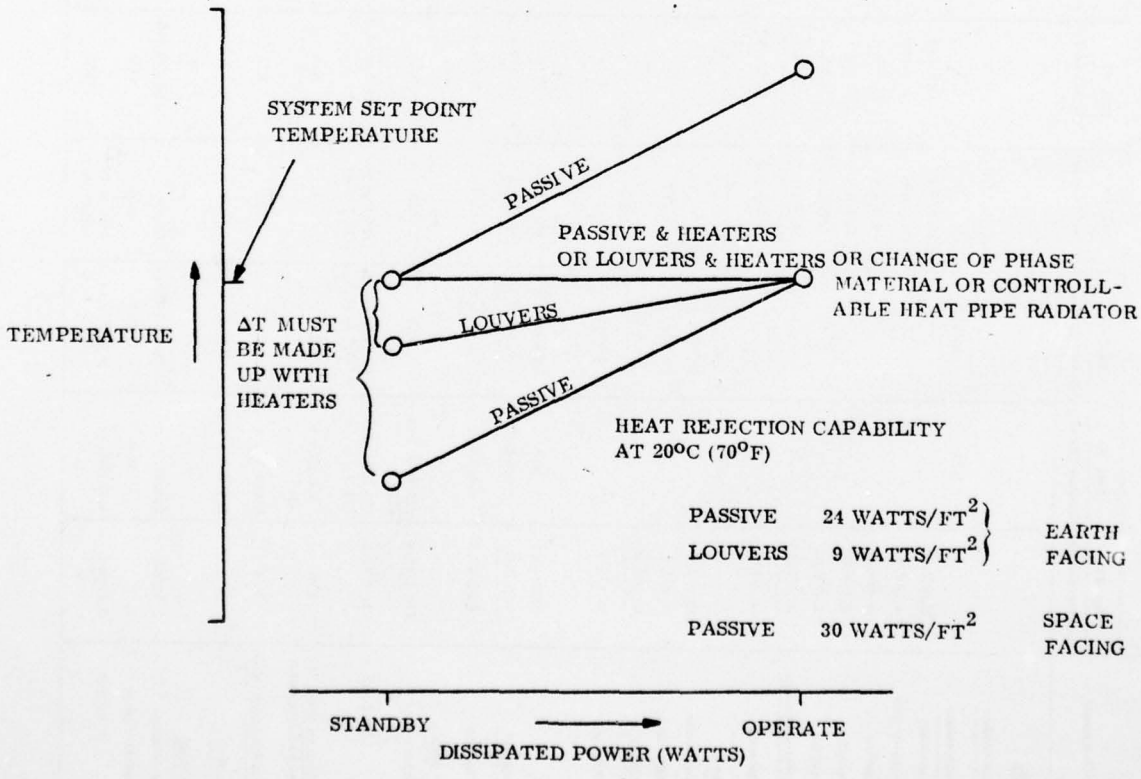


Figure 4.3-1. Thermal Control

Table 4.3-1. Power Dissipation and STR Component Definition

Area/Component	Power Dissipation - Watts										
	Exp. #1	Exp. #2	Exp. #3	Exp. #4	Exp. #5	Exp. #6	Exp. #7	Exp. #8	Exp. #9	Exp. #11	Exp. #12
	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby	Operate/Standby
R Assembly											
Equipment Module											
Communications		N/A	N/A								
Transmitter	13.5/0.0			13.5/0.0	13.5/0.0	13.5/0.0	13.5/0.0	13.5/0.0	13.5/0.0	13.5/0.0	13.5/0.0
Telemetry Unit	8.0/0.0			8.0/0.0	8.0/0.0	8.0/0.0	8.0/0.0	8.0/0.0	8.0/0.0	8.0/0.0	8.0/0.0
Recorder	10.0/0.0			10.0/0.0	10.0/0.0	10.0/0.0	10.0/0.0	10.0/0.0	10.0/0.0	10.0/0.0	10.0/0.0
Receiver	2.0/2.0			2.0/2.0	2.0/2.0	2.0/2.0	2.0/2.0	2.0/2.0	2.0/2.0	2.0/2.0	2.0/2.0
Decrypter	2.2/2.2			2.2/2.2	2.2/2.2	2.2/2.2	2.2/2.2	2.2/2.2	2.2/2.2	2.2/2.2	2.2/2.2
Command Decoder	10.0/4.0			10.0/4.0	10.0/4.0	10.0/4.0	10.0/4.0	10.0/4.0	10.4/4.0	10.4/4.0	10.4/4.0
Subtotal	45.7/8.2			45.7/8.2	45.7/8.2	45.7/8.2	45.7/8.2	45.7/8.2	45.7/8.2	45.7/8.2	45.7/8.2
Payload		N/A		N/A	N/A	N/A	N/A			N/A	N/A
Transmitter	13.5/0.0		N/A					13.5/0.0	13.5/0.0		
Encryptor	7.0/0.0		N/A					7.0/0.0	7.0/0.0		
Recorder	10.0/0.0		10.0/0.0					10.0/0.0	10.0/0.0		
Subtotal	30.5/0.0		10.0/0.0					30.5/0.0	30.5/0.0		
ACS										N/A	
IRU	35.0/35.0	N/A	35.0/35.0	35.0/35.0	N/A	35.0/35.0	35.0/35.0	N/A	N/A		35.0/35.0
Computer	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5	6.5/6.5		6.5/6.5
Subtotal	41.5/41.5	6.5/6.5	41.5/41.5	41.5/41.5	6.5/6.5	41.5/41.5	41.5/41.5	6.5/6.5	6.5/6.5		41.5/41.5
Battery Module											
Batteries - Door	279.4/37.4	233.9/14.1 to 47.6	N/A	N/A	N/A	N/A	N/A	172.9 to 55.7/ 13.7 to 24.8	Not Evaluated	N/A	N/A
- No Door	279.4/63.6	233.9/14.1 to 283.9	187.9 to 112. 8/10.4 to 92.6	121.8 to 99. 3/12.9 to 44.3	189.1 to 48.9	31.8/22.4 to 12.4	27.8 to 18.4	172.9 to 55. 7/13.7 to 51.0	225.9/13.7 to 43.2	23.9/2.0 to 13.2	87.2/49.7
STR Heaters (Control-lage Radiator Only)	N/A	0.0/24.8 to 22.7	N/A	N/A	N/A	N/A	N/A	N/A	N/A	0.10 to 10.0	N/A
Experiment Comp. Mod.	N/A	210/0.0/0.0	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
PM Assembly	N/A	509.0/0.0	N/A	N/A	509.0/40.0	N/A	N/A	509.0/40.0	509.0/40	N/A	N/A
Experiment											
Electronics	1000/100	210/10.0	700 to 400/0.0	300/0	195. to 145.	40./0.	24.0	100./0.0	312./0. to 50.	50.0/0.0	0/0
Heaters - Door	0./0.	0.0/134.0	0.0/39.7	N/A	N/A	N/A	N/A	0/44.4	Not Evaluated	N/A	N/A
- No Door	0./104.7	0.0/1079.	0.0/39.7 to	100. to 10./ 100. to 10.	0/0	0/0	0/0	0/149.4	0.0/0 to 41.5	0.0/0 to 7.1	0/0

Table 4.3-2. Experiment and STR Component Duty Cycle Definition

Experiment No.	Duty Cycle
1	Experiment: 1000 Watts for 30 minutes/orbit for 2 consecutive orbits/day; 100 Watts standby COMM, payload, and ACS: 117.7 Watts; for 15 minutes prior to experiment operation and during experiment operation 117.7 Watts; for 15 minutes 2 times/day during non-operating periods 49.7 Watts standby
2	Experiment: 10% duty cycle or 45 Watt hours/orbit; EPM: 1% Duty cycle
3	Experiment: Operate every orbit only during nighttime (36.4 of 90 minutes): 100 Watts in electronics assumed 50% in transmitter and 50% in power supply; receiver 600 Watts during 1/2 PPS laser converter and 300 Watts during 1/4 PPS laser converter (assume 1/2 each orbit operating period in each mode)
4	Experiment: 300 Watts during each of 10 equally spaced 5 minute operations COMM: 45.7 Watts for 2.5 hours for each operation; 8.2 Watts standby
5	Experiment: Continuous operation at 170 ±25 Watts; telescope 9 Watts internal plus aperature heat load EPM: 1% duty cycle COMM: 45.7 Watts for 20 minutes every orbit; 8.2 Watts standby
6	Experiment: A&B components on continuously at night (36.4 of 90 minutes); A is 40 Watts and B (45 Watts) is experiment peculiar and not included in assessment COMM: 45.7 Watts for 20 minutes every orbit; 8.2 Watts standby
7	Experiment: Continuous 24 Watts operation COMM: 45.7 Watts for 20 minutes every orbit; 8.2 Watts standby
8	Experiment: 10% operation at 100 Watts in areas of interest; 1 hour per operation EPM: 1% duty cycle COMM and Payload: Same duty cycle as experiment for TT&C (45.7 Watts) and payload (30.5 Watts); TT&C 8.2 Watts standby and payload 0.0 Watts standby
9	Experiment: 312 Watts average dissipation during operation with 80 minutes operation maximum per 7 days, 40 minutes operation maximum per day, and 40 minutes maximum operation per orbit; 20 minutes for each experiment operation 15 Watts continuous heater power and 50 Watts heater power for 19 hours prior to each operation EPM: 0.8% duty cycle COMM and Payload: TT&C (45.7 Watts) and payload (30.5 Watts) during operation plus 20 minutes per orbit for all orbits; 2 Watts and 0 Watts standby
11	Experiment: 50 Watts during each of 10-15 minute encounters - maximum of 4 encounters per day Assume 50 Watts equally distributed between 3 radiometers COMM: 45.2 Watts for 20 minutes per orbit; 8.2 Watts standby
12	Experiment: Autonomous experiment operation COMM: 45.7 Watts for 20 minutes once per day; 8.2 watts during standby

4.3.2 DESIGN TRADE STUDIES

The STR is completely enclosed in multi-layer insulation. Heat pipes, located on the module panels and STR structure, were selected to distribute the STR heat dissipation. Three heat rejection concepts were evaluated using the Shuttle liquid loop, phase change, or controllable heat pipe radiators. (Drawing STR003, Sheet 3 in Appendix E). The power and thermal requirements for these concepts were established by an evaluation of the requirements for each experiment. For those experiments requiring an EPM, an independent thermal control concept was utilized for the EPM. The thermal control approach for each experiment and the EPM is described in Appendix B.

It is concluded that a thermal door (also contamination protection) should be provided for all optical experiments to minimize power usage and thermal control requirements. The STR thermal requirements summary by experiment are presented in Table 4.3-3 for each candidate heat rejection system. The advantages and disadvantages of each heat rejection concept is shown on Table 4.3-4. The design definition and tradeoffs involved with each heat rejection concept are described in the following sections.

Standard flight proven thermal control components have been selected for use except for the phase change module concept for heat rejection. The design requirements and preliminary definition for three heat rejection concepts (Shuttle liquid loop, phase change, controllable heat pipe radiator) have been established, along with their advantages. A more detailed design and development test is recommended for the total concept selected.

4.3.2.1 Shuttle Liquid Loop

Potential Shuttle liquid loop interfaces are defined on Figure 4.3-1. As is indicated in Figure 4.3-1, the use of the Shuttle loop is dependent on the total orbiter load and configuration. If the STR must use a secondary loop, the 30°C maximum STR temperature cannot be maintained. For higher allowable STR temperatures of 35°C, 45°C and 55°C, the STR may be used with Spacelab, IGLO and SSPE pallet, and spacelab plus Pallet, respectively. The maximum STR instantaneous heat load is only 500 Watts and is well within the 8.5 kW orbiter capability even with other orbiter heat loads.

Table 4.3-3. STR Thermal Requirements Summary

			Heat Rejection Approach								
			Phase Change (PC)			Controllable Radiator (CR)				Liquid Loop (LL)	
Exp. No.	Experiment Name	Electrical Energy KWH	Energy KWH	No. STR Sides	On Pad Time After -20°C Sink Removed Hours	Max Load Watts	Area ft^2	Size "X"	No. STR Sides	Max Load (Watts)	No. STR Sides
#1	Laser Com					279.4	21.83	66.4 x 47.3			
	Receiver Test Door	32.1	17.1(9.0/8.1)	2	23.7	117.7	9.19	28 x 47.3	2	397.1	1
						279.4	21.83	66.4 x 47.3			
	No Door	44.2	23.1(9.0/12.3)	2	-	117.7	9.19	28 x 47.3	2	397.1	1
#2	Hirise Door	25.2 (PC&LL)/				233.9	18.27	55.6 x 47.3			
		28.8 (CR)	9.2	1	23.2	210	16.4	49.9 x 47.3	2	443.9	1
		167.3 (PC&LL)/				283.9	22.17	67.5 x 47.3			
	No Door	170.7 (CR)	46.4(6.9/39.5)	2	-	210	16.4	49.9 x 47.3	2	493.9	1
#3	Atmospheric Topside										
	Laser Sounder (Atlas)	40.1	19.4(7.7/11.7)	2	16.0	239.4	18.7	56.9 x 47.3	1	229.4	1
#4	Precision Release										
	Accuracy Test	18.3	13.9(9.3/4.6)	2	22.9	209	16.33	49.7 x 47.3	1	209	1
#5	Earth Limb										
	Measurements Sensor	44.1	14.8(3.9/10.9)	2	18.3	241.3	18.85	57.4 x 47.3	1	241.3	1
#6	LASS II	12.0	12.7(9.7/3.0)	2	21.7	119	9.30	28.3 x 47.3	1	119	1
#7	XUV	13.8	13.2(9.7/3.2)	2	21.7	115	8.98	27.4 x 47.3	1	115	1
#8	Optical Geophysics and										
	Astronomical Door	18.6	8.1	1	26.3	255.6	19.97	60.8 x 47.3	1	255.6	1
	Observatory No Door	32.1	11.5	1	-	255.6	19.97	60.8 x 47.3	1	255.6	1
#9	Teal Ruby	18.8	9.8	1	21.4	308.6	24.10	73.4 x 47.3	1	308.6	1
#11	Optical Counter										
	Measures Demonstration	3.9(PC & LL)/ 5.6 (CR)	3.8	1	38.6	69.6	5.44	16.6 x 47.3	1	69.6	1
#12	Deployable										
	Spacecraft	8.5	10.7	1	18.9	109.0	8.52	25.9 x 47.3	1	109	1

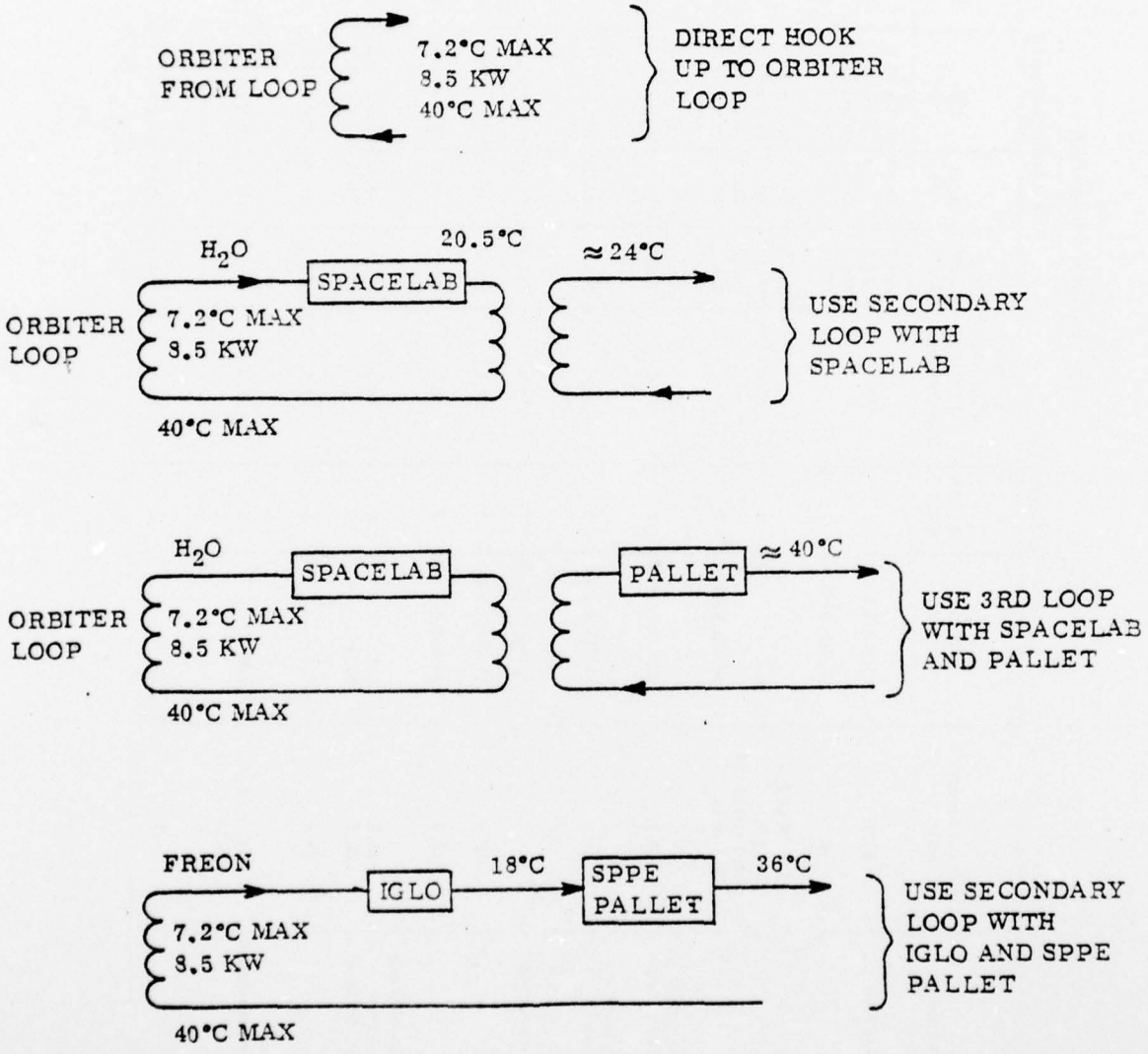


Figure 4.3-1. Shuttle Liquid Loop Interfaces

Table 4.3-4. Heat Rejection Concept Comparison

Concept	Advantages	Disadvantages
Shuttle Liquid Loop	Light Weight	Has limited use based on Shuttle primary payloads
Phase Change	Minimum Shuttle/experiment interface may be used for all experiment	Heaviest approach limited heat rejection capability
Controllable Heat Pipe Radiator	Lightest Weight	View factors to shuttle and experiment need to be evaluated. Experiment F.O.V. may make use of radiator difficult

However, the 30°C STR temperature can only be maintained if the upstream orbiter heat loads result in a maximum inlet water temperature below 16°C. The baseline orbiter loop provides 8.5 kW of cooling with a maximum inlet temperature of 7.2°C. With 500 pounds per hour of water coolant flow at 65 PSI, the cold plate design shown on Drawing STR 003 Sheet 3 in Appendix E consisting of 23.3 feet of 1/2 inch tubing thermally integral with a 37.2 inch by 43 inch plate fastened to one module panel, maintains a maximum STR temperature of 21°C. The cold plate pressure drop is 0.23 psi and the weight of the water, lines, plate, and pump assembly will be about 150 pounds.

4.3.2.2 Controllable Heat Pipe Radiator

A controllable heat pipe radiator can be located on one or both sides of the STR to reject the STR heat dissipation. The radiator, which must be integrated with the experiment (to allow clearance and fields-of-view), consists of 4 controllable (2 each end) heat pipes which are thermally integral with the STR distribution heat pipes and extend along the edges of the radiator plate. Four fixed heat pipes are used to distribute the heat over an 0.06 inch Aluminum plate. The radiator thermal control coating is 5 mil Teflon over

Silver with a degraded a/ϵ of 0.12/.78. The configuration is shown on Drawing STR003 sht 3 in Appendix E. With the sun normal to the radiator, the heat rejection capability is 12.8 Watts/ft^2 at 15°C and, based on the maximum STR heat load for each experiment, the radiator sizes defined in Table 4.3-3 were established. At 15°C ΔT between the STR module and radiator was assumed. Some experiments require a radiator on both sides of the module while most experiments can package and reject their heat from one side. The minimum heat load required to hold 10°C is 22.7 Watts per STR side and STR heaters would provide this load whenever the STR dissipation is below 22.7 Watts. For maximum STR temperatures of 35°C, and 45°C and 55°C the radiator size could be reduced 14%, 33% and 46%, respectively. Assuming a maximum radiator size of 70 by 47.3 inches, the weight of the radiator assembly including four fixed heat pipes, four controllable heat pipes and the plate is about 40 pounds (one STR side).

4.3.2.3 Phase Change

A phase change module, mounted to one or more STR modules, can provide heat rejection via energy storage using heat of fusion. The heat load requirements for each experiment are shown on Table 4.3-3. From the data a minimum module capable of 11.7 kWh for STR operation is required. The phase change material selected was ice/water. Since ice has a 0°C melting point and there will be unattended pad operation for 15 hours normally in a 25°C environment, the phase change module size must be adequate to accommodate the pad tare heat load. Trade-studies, shown below, indicated that ice was preferable over other phase change materials with higher melting points. Accommodating the pad tare load for 15 hours with a phase change unit which physically fits within the current module space envelope requires the pad cooling system to be designed to maintain the module at -20°C rather than 0°C on the pad. This will result in STR component temperatures approaching -20°C during non-operation on the pad. Using the thermal time constant of the module, it will take 5.6 hours for the module to increase in temperature from -20°C to 0°C and the pad tare load is decreased to 3.3 kWh which results in a phase module designed for 15.0 kWh (11.7 + 3.3 kWh). The module configuration, shown on Drawing STR003 sht 3 in Appendix E, has a 39 inch baseplate with a 36.6 by 40.7 inches phase change volume with a 4 by 4 by 7 inches high aluminum metal structure internal matrix to transfer heat more evenly to the phase

change material. The phase change module baseplate thermally interfaces with the STR heat pipe distribution system and the phase change module top has integral coolant lines to interface with the pad cooling system. The assembly has a 15.35 kWh capability and the resulting pad hold times for each experiment after the -20°C coolant is removed is shown in Table 4-16. The unit weighs 443.8 pounds. The areas in this design requiring some development are material containment and conduction to the unmelted material in a 0 g environment. It has the advantage of complete independence from both Shuttle and Experiment interfaces. A phase change concept is therefore attractive for short term missions. Each module is capable of a total of 15.35 kWh energy usage. When this is exceeded, the STR/Experiment temperature will increase until termination of experiment operation. Under direct sun load in orbit with no internal power dissipation, the thermal sink temperature is -25°C so that no additional orbit tare load is present for the phase change module. In fact, the tendency is to refreeze the water.

The first phase change trade off made was to determine the ΔT across the phase change material as a function of internal module matrix size. A 21 node thermal model of a 1 x 1 inch, 2 x 2 inches, and 4 x 4 inches matrix was established. The ΔT across an 8 inches high cell was determined for water as shown in Figure 4.3-2. With T's of 1.8°C. 4.2°C, and 8.5°C respectively, a maximum module base temperature was established for the orbital design. There is an assumed 10°C ΔT between the STR equipment modules and phase change module baseplate with a maximum heat load of 300 watts. With a 30°C Equipment/module, a 20°C temperature is held at the phase change module base. The maximum heat load from the phase change module panel to the phase changing module is 500 Watts. The matrix ΔT, in conjunction with a 20°C required module interface, defines the maximum thermal isolation allowable at the phase change module/ STR structure interface. A 24 node thermal analytical model of the STR in the ground environment was established to determine the tare heat load on the phase change module. For a 4 x 4 matrix the heat load is 350.1 Watts. A comparison of other matrix sizes, other phase change materials, and higher STR temperatures was conducted as summarized in Table 4.3-5. On the basis of available volume, minimal weight, and adequate temperature control, a 4 x 4 matrix with ice maintaining a maximum 30°C

Table 4.3-5. Phase Change Material Comparison

	Material Properties				Orbit Temperature Performance					Phase Change Material Requirements					
Material	Melting Point	Heat of Fusion BTU/#	Density #/ft^3	Thermal Conductivity BTU/hr H°F	Matrix Size Inch = Inch	Matrix T °C	PCM Baseplate Temp. °C	STR Structure Temp. at PCM °C	Equipment Panel Temp. -°C	PAD Heat Load Watts	PAD Energy For 15 Hours KWH	Total PCM Energy Req'd KWH	PCM Weight #	PCM Volume ft^3	PCM Size W" x L" x H"
Eicosane ($C_{10}H_{42}$)	36.6	106	48.7	.086	1 x 1	7.0	43.6	44.6	54.6	-	0	11.7	376.7	7.74	43.7 x 43.7 x 7
Octadecane ($C_{28}H_{38}$)	27.7	105	48.5	.087	1 x 1	7.0	34.7	35.7	45.7	-	0	11.7	380.3	7.84	44.0 x 44.0 x 7
Hexadecane ($C_{16}H_{34}$)	17.7	102	48.3	.086	1 x 1	7.0	24.7	25.7	35.7	140.7	2.1	13.8	461.8	9.56	48.6 x 48.6 x 7
Tetradecane ($C_{14}H_{30}$)	5.5	98	48.3	.086	1 x 1	7.0	12.5	20.0	30	303.4	4.6	16.3	567.7	11.75	53.9 x 53.9 x 7
Ice/Water	0.0	144	62.4	.335	1 x 1	1.8	1.8	26.0	30	302.8	4.5	16.2	379.2	6.08	38.7 x 38.7 x 7
					1 x 1	1.8	1.8	25.0	35	274.8	4.1				
					1 x 1	1.8	1.8	30.0	40	251.9	3.8				
					2 x 2	4.2	4.2	20.0	30.0	317.0	4.8				
					2 x 2	4.2	4.2	25.0	35.0	286.2	4.3				
					2 x 2	4.2	4.2	35.0	45.0	210.9	3.0				
					4 x 4	8.5	8.5	20.0	30	350.1	5.3				

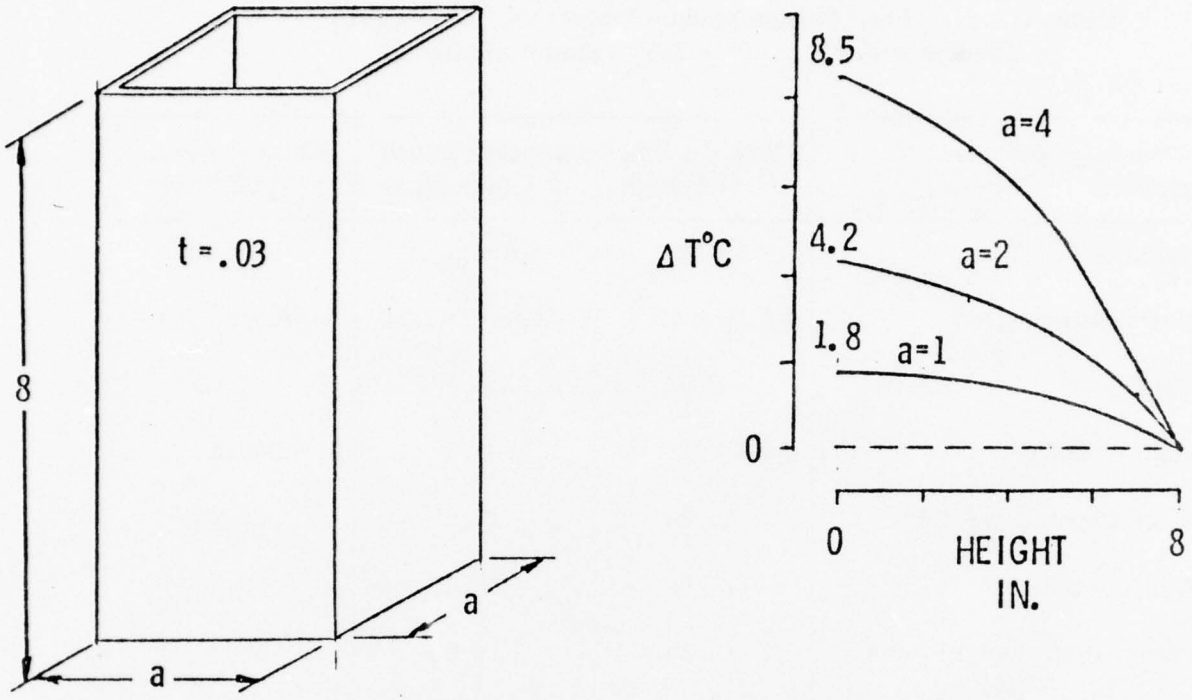


Figure 4.3-2. Change of Phase Module Design (Preliminary)

equipment panel temperature was selected along with a pad cooling system which maintains the phase change module at -20°C until at least 15 hours prior to launch. The weight of other matrix sizes is shown on Table 4.3-6. Decreasing the matrix size would improve thermal confidence at the expense of some additional weight.

4.3.3 THERMAL COMPONENTS

4.3.3.1 Thermal Coatings

Thermal coatings are used on external surfaces to control the absorbed and reflected heat fluxes, the radiative coupling between vehicle surfaces, and the heat radiated from external vehicle surfaces. Typical thermal control coatings are presented in Table 4.3-7, along with their normal usage and optical properties at both beginning of mission (BOM), and end of a 10 year mission (EOM). Coatings selected for the STR are indicated.

Table 4.3-6. Phase Change Module Weight vs. Matrix Size
(Water/Ice with 39" x 43" x 7.5" Volume Available)

Parameter \ Matrix Size	1 Inch x 1 Inch x 7 Inch High	2 Inch x 2 Inch x 7 Inch High	4 Inch x 4 Inch x 7 Inch High
No. of Matrices	35 x 39	18 x 20	9 x 10
PCM + Matrix Dimensions	37.16 x 41.4	37.14 x 41.26	36.6 x 40.7
PCM Volume - ft^3	5.53	5.83	5.83
PCM Weight - Pounds	345.1	363.8	363.8
PCM Energy Capability - KWH	14.56	15.35	15.35
Matrix Weight - Pounds	117.96	63.77	33.48
Top & Bottom Plate Weight - lbs	19.29	19.26	19.0
Tubing & Misc Wt. - lbs	27.75	27.0	27.52
Total Module Wt. - lbs	510.1	473.8	443.8
Energy to Heat PCM 20°C - BTU	6965.0	6882.0	6666.0
PAD Heat Leak to °C PCM - Watts	302.8	317.0	350.1
Time to Heat PCM 20°C - hrs	6.74	6.36	5.58
PAD PCM Tare Load for Remainder of 15 Hour PAD Time with -20°C Sink Removed - KWH	2.5	2.74	3.30
PCM Total Module Capability Required with 11.7 KwH Experiment Load - KWH	14.2	14.44	15.0

The heat rejection coating selected is 5 Mil Teflon over Silver. This coating has acceptable thermal properties with considerable flight performance data as shown on Table 4.3-8. The STR evaluation utilized a maximum expected degraded s/e ratio for one year of 0.12/.78. The bonding material for this coating is GESR585. The silver Teflon coating will be grounded at local tabs.

AD-A064 773 GENERAL ELECTRIC CO PHILADELPHIA PA SPACE DIV F/G 22/2
STANDARD TEST RACK EXPLOITATION STUDY.(U)
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UNCLASSIFIED SAMSO-TR-77-190 NL

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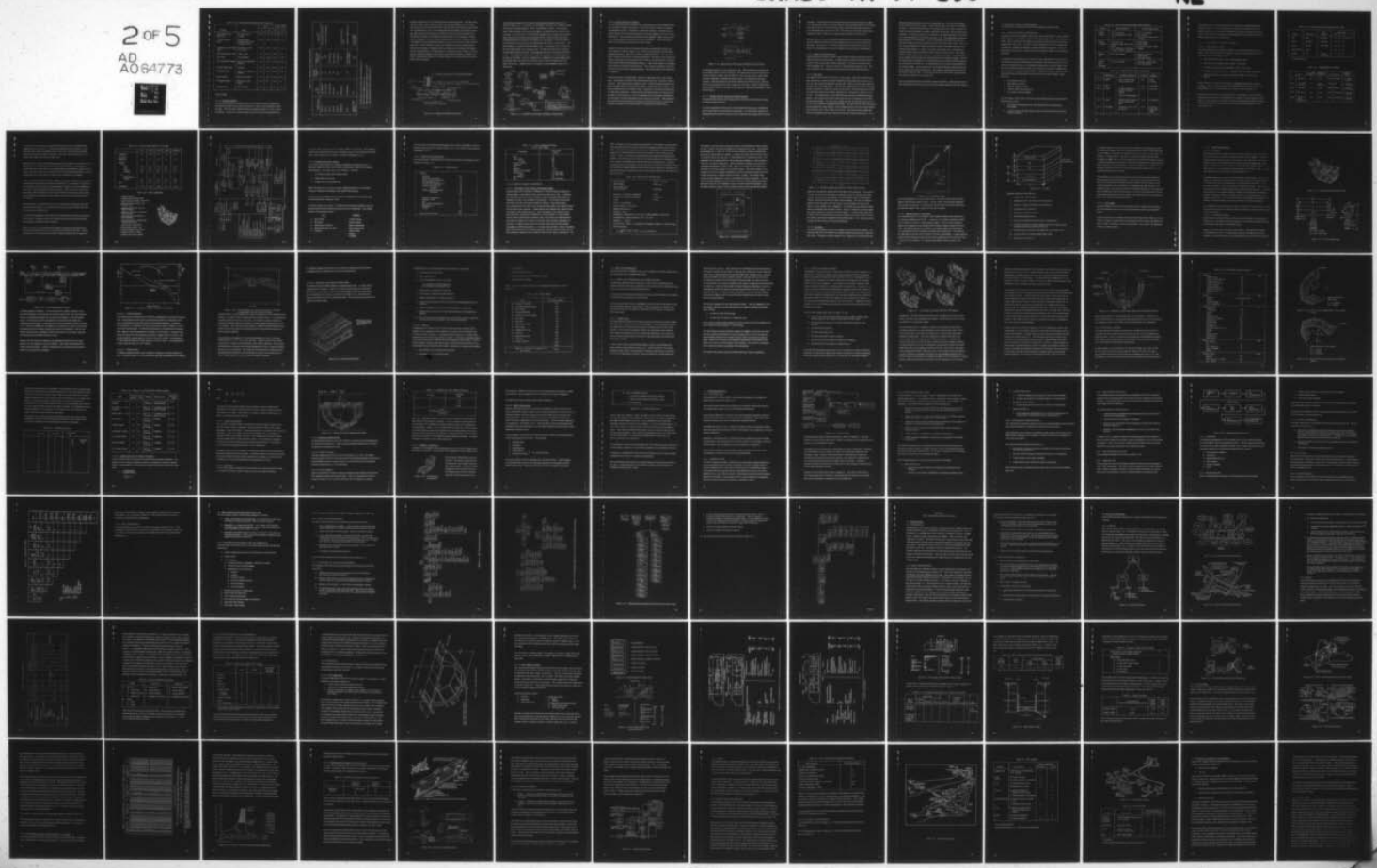


Table 4.3-7. Thermal Control Coatings Optical Properties

Coating	Usage	BOM		10 Year EOM	
		α_S	ϵ_H	α_S	ϵ_H
OSR (Optical Solar Reflector)	Heat Rejection Coating	0.08	0.80	0.24	0.80
*5 Mil Teflon over Silver	Heat Rejection Coating - outer layer insulation blanket	0.08	0.78	0.26	0.78
*Chemglaze Z-306 Black Paint	S/C Interior, Solar Array Rear	0.95	0.88	0.95	0.88
Aluminized Kapton Tape	Tanks, Lines	N/A	0.05	N/A	0.05
Gold Plate	Nozzle Shields	0.10	0.30	0.10	0.30
2 Mil Aluminized Kapton	Outer Layer Insulation Blanket	0.40	0.72	0.56	0.72
S-13GLO White Paint	Antennas	0.21	0.82	0.50	0.82
Fiberglass Cloth	Outer Layer Insulation Blanket	0.25	0.74	0.60	0.74
Black Anodize	Solar Array Shaft	N/A	0.80	N/A	0.80
*Aluminized Mylar	Insulation, Lines, Tanks	N/A	0.5	N/A	0.5
Aluminum Foil	AM Membrane	N/A	0.5	N/A	0.5

*STR Coatings

4.3.3.2 Insulation Blankets

The multi-layer insulation blankets provide very low effective thermal conductivity, isolating the satellite from the space environment. They are also grounded for space charging. The multi-layer insulation blanket is depicted in Figure 4.3-3 for a typical installation. An effective emissivity range from 0.005 to 0.020 is considered with a

Table 4.3-8. Metallized Teflon Flight Performance

Flight Program	Orbit (n. mi.)	Minimum Temperature (°C)	Change In Solar Absorptance	Remarks[1]
OAO-C	400 Equatorial	- 28	.06	2000 Equivalent Sun Hours
IMP-I	170 x 110,000	- 71	.12[2]	2 Years in Orbit Spinning
SAS-B	400 Equatorial	-100	None Apparent	
OGO-6	270 x 540	- 50	None Apparent	
OSO-H	300	N/A	None Apparent	
Intelsat IV	Synchronous	-	.08	2 Mil Al Teflon after 4 years
CTS	Synchronous	-110	None Apparent in first 4 Months	Conductive Adhesive Used
Mariner Venus Mercury	N/A	-	.14 .25	19,000 ESH[3] 31,000 ESH[4]

1. All data refers to silver coated Teflon except Intelsat IV.
2. Prior to 6 hour eclipse Δa was .08.
3. Total charged particle fluence estimated at 6×10^{16}.
4. Total charged particle fluence estimated at 10^{17}.

nominal analysis value of 0.015 which has been verified by test data. The outer 5 mil Teflon over silver layer provides an outer covering for handling, a low α/ϵ ratio, and protection for the inner blanket layers from the space environment for the mission. The remaining layers are 1/4 mil aluminized (both sides) Mylar, perforated for venting, with Dacron mesh separators to provide the required low thermal conductivity. The blanket assembly uses fiberglass threading on about 12 inch spacing to maintain blanket configuration with Velcro hook and pile fasteners for ease in blanket removal. A few higher load carrying insulation buttons serve as close tolerance blanket positioning devices. Special attention to edge seals at blanket to thermal coating interfaces minimizes the trapping of sunlight in these areas and prevents venting of the blankets onto the coating surfaces. All blankets have a minimum of 3 inch overlap, or will have separate collar type blankets in place of butt type, blanket interfaces. Venting is provided by standard perforations of 0.050 inch diameter on 3.0 inch centers in the aluminized Mylar and Kapton.

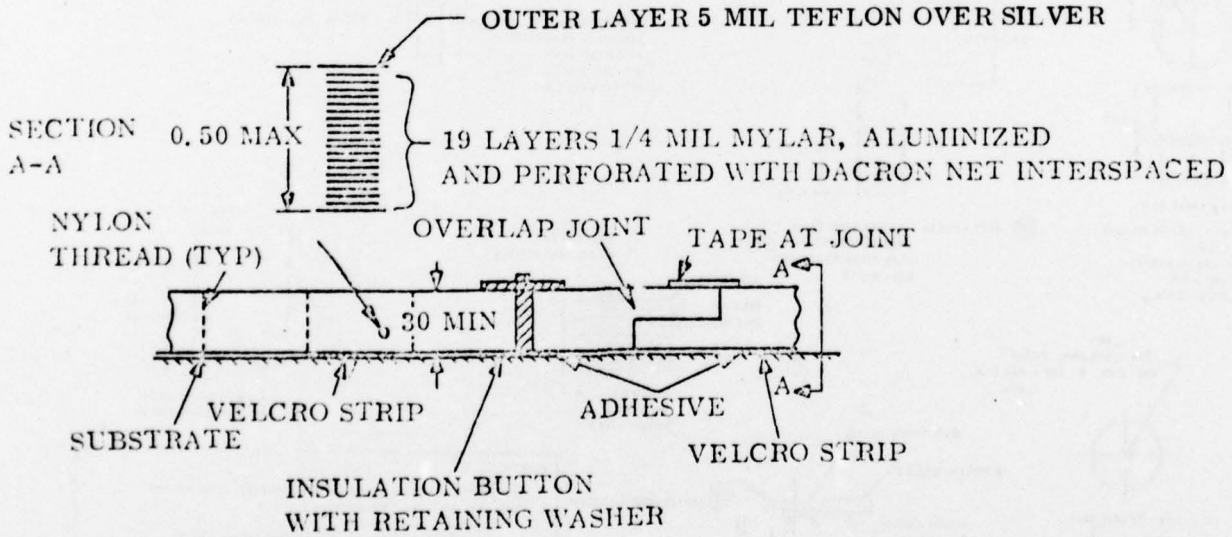


Figure 4.3-3. Insulation Blanket Cross-Section

The multi-layer insulation EMI protection and grounding technique used by GE is shown in Figure 4.3-4. The blankets are grounded by "shorting" the metallized surface of each layer to satellite ground. This is accomplished as follows. The alternate layers of dacron mesh are cut away to the edge (a vee notch is used for ease of manufacturing). A one inch wide, 0.001 inch thick aluminum strap is folded in an accordian shape such that it will contact each metallized surface. This accordian fold is then compressed between two washers held together by a nut and bolt. One of the washers is metal and has a raised lip to ensure high compression forces at the periphery. The metal washer is in contact with the aluminum strap. An aluminum wire is attached to the metal washer (under the retaining screw). The other end of this wire is attached to the satellite structure (ground). This design has been proven in rigorous ESD tests on BSE. The specification requirement is that the resistance to satellite ground from any layer be $\leq$10 ohms. There is a minimum (for redundancy) of two grounding straps for blankets under one square foot and three or more straps for blankets over one square foot in area. Typically there is a minimum of one grounding strap per 2 to 3 square feet.

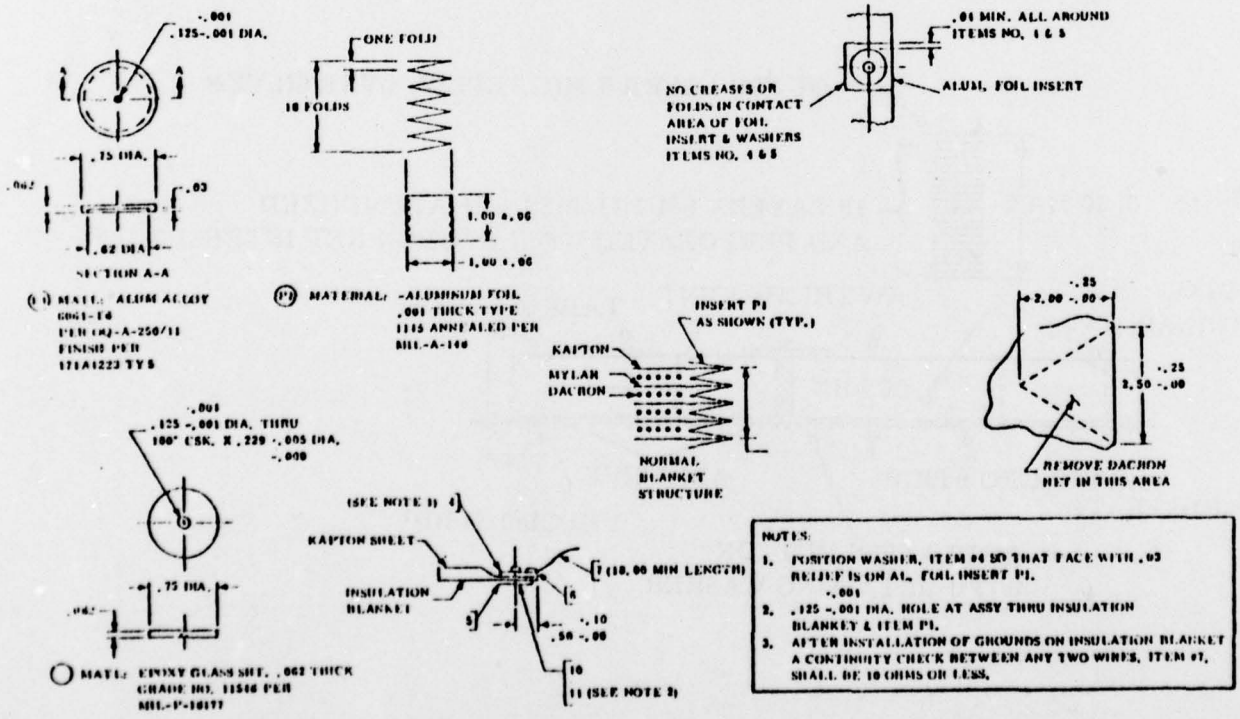


Figure 4.3-4. Insulation EMI and Space Charging Grounding Design

4.3.3.3 Heater/Thermostat Assembly

The TCS heater/thermostat assemblies have redundant heaters and redundant thermostats to eliminate single point failures. These assemblies are located as required for temperature control. Each heater is internally redundant. Each thermostat assembly consists of two control thermostats, to provide redundancy, and two overheat thermostats. The circuits have command override capability. Both A and B circuits are command engaged simultaneously to ensure redundancy. A flight temperature sensor is located at each thermostat assembly location.

A typical electrical circuit for the heater/thermostat assembly (both parallel and series heaters) is shown in Figure 4.3-5. This circuit assumes two control and two overheat thermostats. The circuit contains no single point failure. The "A" and "B" heater circuits are not tied to a common ground point. Any heater circuit which would require a thermostat to switch greater than 28 Watts at 28 Volts will have a transistor switch located between the thermostat and heater. This switch consists of a darlington circuit which is controlled by the thermostat. This concept was qualified on BSE and is used for EMC and current derating purposes. The rated capacity of the thermostats is 4 amperes.

The heaters selected are lightweight, etched foil, Kapton film heaters with welded leads. The heaters are bonded to the substrate with Hysol 956. The heater vendor is Tayco Inc. This design has been qualified on BSE. Heater/thermostat assembly operation is verified at several point in the acceptance, installation, and test cycle. Each heater and thermostate is acceptance tested. Operation of each heater and thermostat is verified after installation. During satellite acceptance testing, the primary circuit operation is verified (primary circuit on-redundant circuit off) and the redundant circuit operation is verified (redundant circuit-on-primary circuit off). This testing will prove that the control thermostat is working, the overheat thermostat is closed, and that there is adequate circuit heater power (via temperature telemetry data).

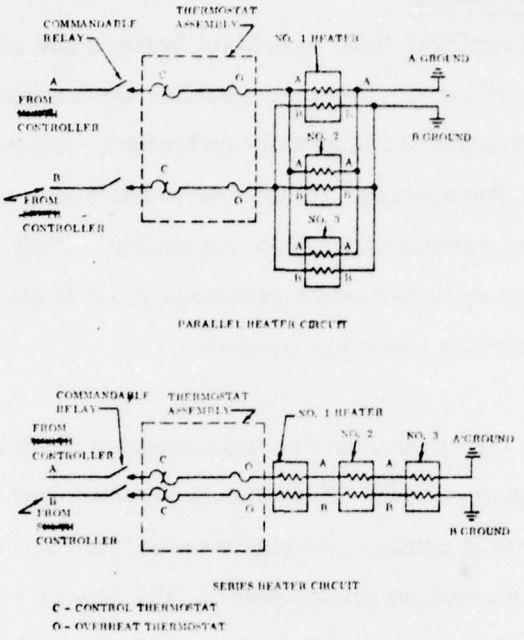


Figure 4.3-5. Typical Heater/Thermostat Assembly Electrical Circuit

The thermostat vendor is Texas Instruments, Inc. This thermostat is the vendor M2 series which conforms to MIL-S-24236/20. The thermostats have been used by GE on military programs, Skylab, BSE and DSCS III. They are also used by various other Aerospace companies, including McDonnell Douglas, Air instruments and Martin Marietta. These thermostats have been life tested by the vendor for up to 1,500,000 cycles on one particular device with a maximum 3.9°C drift. At least 27 different life tests have been completed including 250,000 cycles on a thermostat operating at 2 amps and 30 Volts dc.

4.3.3.4 Thermal Interface Material and Isolation Spacers

Thermal interface material and isolation spacers are used to control thermal conduction between vehicle surfaces.

For many components the temperature requirements and internal power dissipations make a metal/metal mounting interface unacceptable. An electrically conductive low outgassing thermal interface material is used to minimize the mounting contact thermal

resistance. The interface material reduces the effects of flatness tolerance, roughness, and non-uniformities in the panel substate and component baseplate. The interface material selected is RTV-566 with silver filler, which has outgassing properties of 0.1% total weight loss, and 0.01% CVCM. If the electrical resistance requirements, component to panel, cannot be met, then a separate ground wire will be used.

Isolation spacers are used at component mounting interfaces to meet critical local thermal conductance requirements not obtainable with basic spacecraft mechanical interfaces. The spacer is a low thermal conductivity Textolite material that is designed for each application to meet the combined structural/thermal requirements.

4.3.3.5 Flight Temperature Sensors

Flight temperature sensors are used to verify satellite in-flight performance and for diagnostic/operational information. A flight sensor is located at each thermostat assembly as well as many other satellite components. All sensors are analog at the slowest sampling rate. The sensor vendor is Yellowsprings Instruments, Inc. This type of thermistor, part number R4221, has been used on many flight programs.

4.3.3.6 Heat Pipes

Heat pipes are used on the STR structure, module panels, (and with controllable heat pipe radiator heat rejection concept) to isothermolize the panels and structure to provide the required heat distribution to the heat sink. There are 18 heat pipes required per STR side in two configurations. The heat pipe configurations are shown in Drawing STR001 in Appendix E along with a typical cross-section. The heat pipes must provide up to 2000 Watt-inches heat transport capability over a temperature range from -15 C to +55 C. The design selected is the 6063 aluminum axial groove tubing with integral flanges. Increased thermal contact conductance and ease of installation are provided by the integral flanges. The axial groove design has been successfully proven on OAO-C, ATS-6, IEU and GSFC Sounding Rocket flights. The tube vendor is Micro-Extrusions, who has supplied heat pipe tubing in various configurations for several heat pipe applications including ATS-5, BSE, Skylab, IUE and PAC. The internal groove design with 27 internal longitudinal grooves is identical to the GSFC standard axial groove. Ultra

high purity ammonia will be used as the working fluid, and is the best fluid available for the required temperature range. (Argon would be used as inert gas in controllable heatpipes). The thermal performance of an ammonia/groove heat pipe is measured in terms of axial heat transport capability, radial heat flux density, and total temperature gradients. All of these parameters are functions of operating temperature, percentage fill, non-condensible gas contaminants and tilt (in 1-G). The compatibility of ammonia with aluminum is well established. GE has considerable experience with ammonia/aluminum designs and directly applicable life test data from the BSE design. The axial heat transport capability is usually measured in terms of Watt-inches and may be interpreted as the ability to transport a given heat load (Watts) over a measured axial distance (inches) without exceeding the inherent limits of the heat pipe. These limits include capillary pumping (hydrodynamic), entrainment, and sonic. For the ammonia groove design, the heat transport is always limited by the capillary pumping. The STR heat pipe transport capability will be demonstrated to be in excess of 3500 Watt-inches for each pipe at a 0.10 inch tilt. The radial heat flux will not limit the performance of the proposed design. Tests performed for ATS-6 showed that the radial heat flux limit was in excess of 50 watts per linear inch. The maximum radial heat flux for STR will not exceed 4 Watts per inch. For this radial heat flux the temperature gradient within the evaporator region will not exceed 2.0°C.

Control of manufacturing and processing, hardware inspection, verification and acceptance testing will utilize the same procedures successfully employed on the BSE program.

4.4 ATTITUDE CONTROL SUBSYSTEM (ACS)

The STR must provide attitude determination and payload pointing services.

4.4.1 ATTITUDE CONTROL REQUIREMENTS

An analysis of the twelve candidate experiments indicates that the STS on board attitude control system is not adequate to meet all of the experiments' needs. (See Table 4.4-1.) HIRISE, ELMS, OGAO, and Teal Ruby need a pointing system to satisfy their slewing requirements. (See Table 4.4-2.) PRAT needs a pointing system for its accurate rate requirements. (Although not indicated in Drawing STR 005 in Appendix E) The other experiments attitude control requirements can be satisfied by shuttle except the attitude knowledge.

To satisfy these requirements a more accurate means of pointing must be provided for some payloads in order to operate them in the Shuttle. Each of these payloads can provide its own pointing system or a standardized pointing system can be provided. Over the years of Shuttle operation it will be more cost effective to provide a standardized system which has a standard payload interface. Along with this standardization for the payload (reducing individual payload cost) there are several other benefits. Among these are:

1. High degree of autonomy
2. High degree of security
3. Minimum shuttle interface
4. Reduced shuttle integration time
5. Versatility of payload handling.

The STR attitude control system which provides the pointing and knowledge consists of the following two parts:

1. An attitude determination system with attitude sensors and integrating electronics.
2. A gimbal system with stable platform and drive electronics which forms the pointing mechanism.

Table 4.4-1. Payload Requirement versus Shuttle Capability

Function	Shuttle Capability	Payload Requirement
Attitude Pointing	0.5° (3 σ at IMU) ≈2° Misalignment in Cargo Bay	0.5° ± 0.1° (HIRISE) ±0.4° (ELMS) ±0.5° or more (Other Experiments)
Attitude Knowledge	0.073° Read out (Optimistic since other errors exist) ±0.1° Deadband, ±0.133° alignment, 0.105°/hr. drift rate	0.033° (HIRSE) 0.023° (ELMS) 0.10°or more (other experiments)
Attitude Rate	0.01°/Sec (vernier)	0.005°/sec (OGAO) 0.033° (ELMS) 0.005°/sec (PRAT)
Attitude Rate Knowledge	0.00175°/sec drift	0.006°/sec (LASER COM) 0.001°/sec (PRAT)

Table 4.4-2. Attitude Control Pointing Requirements

	Experiment	Pointing Requirement	Accuracy	Stability
No. 2	HIRISE	50 nm on either side of the Ground Track	0.5°	0.033° 0.1°/sec.
No. 4	PRAT	—	0.5°	0.005°/sec.
No. 5	ELMS	In orbital plane, from horizon to Nadir to horizon ±30° out of orbital plane	0.4°	0.01°/sec.
No. 8	OGAO	Nadir to horizon scan in and out of the orbital plane	1.0°	0.005°/sec.
No. 9	Teal Ruby	Pitch from -60° to +80° on roll ±10°	0.7°	0.01°/sec. 0.004°/sec. jitter

The pointing system is modular and provides a wide range of capability when integrated with the STR. Software packages are required on a modular basis to meet individual payload needs. With a "STR Payload Users Guide" for payload integration and the choice of software packages, a Mission Manager can select the proper mix of hardware and software for each payload.

4.4.2 DESIGN TRADES AND DESIGN SUMMARY

4.4.2.1 Payload Pointing System

Five pointing mechanisms studied are capable of meeting STR experiment requirements. These are listed below with sponsors. Status, source and relative costs for each system are shown in Table 4.4-3.

1. Instrument Pointing System (IPS) - European Space Agency
2. Small Instrument Pointing System (SIPS) - NASA MSFC
3. Annular Suspension and Pointing System (ASPS) - NASA GSFC
4. Miniature Pointing Mount (MPM) - (Appollo Telescope Mount) - No Sponsor
5. Payload Orientation and Instrument Tracker for Shuttle (POINTS) - No Sponsor

These systems do not include attitude sensors, integrating electronics or payload mounting. These elements will be part of the attitude determination system discussed in paragraph 4.4.2.2. The capability of the systems is shown in Table 4.4-4.

Each of the pointing systems could be utilized for all experiments except the HIRISE. Only the IPS and MPM have the weight and size capability to accommodate that payload. However, if the cryogenics can be divorced from the instruments, the gimbal mounting weight for HIRISE would be approximately 600 KG and could be handled by the ASPS and POINTS as well.

Table 4.4-3. Candidate Experiment Pointing Mounts (EPM)

Mount	Fabricator	Date Available	Costs, M$	
			NR	R
IPS	ESA	Mid 1980	4.8*	1.7* (5 to 8)
SIPS	Ball Bros	Mid 1980	2.5	1.0
ASPS	Sperry	Early 1981	1.6	1.0
MPM (ATM)	MSFC		0.8	0.5
Points	GE	2 Years Lead Time	—	0.8

*Estimated by JPL

Table 4.4-4. Pointing System Capability

	Type	P/L Weight Capability, KG	Payload Size Capability, M	FOV	Isolation	Accuracy & Stability
IPS	End Mount	2000	Large	HEMI	Mechanical	P 2 Arc Sec S 1 Arc Sec
SIPS	C.G. Mount Dual Yoke	600	1x1x3 Each Yoke	HEMI	None	P 1 Arc Sec S 0.1 Arc Sec
ASPS	End Mount	600	Large	HEMI	Magnetic	P ~ 1 Arc Sec S 0.01 Arc Sec
MPM	End Mount	1000	Large	HEMI	Mechanical	P 2.5 Arc Sec S 1 Arc Sec
Points	End or C.G. Mount Dual Mounts	680	2xAxB	HEMI	Mechanical	P 1 Arc Sec S 1 Arc Sec

A Life Cycle cost analysis was performed with a basic mission of 4 pointing system flights per year for ten years. Utilizing the acquisition costs from Table 4.4-3 and data from usage studies of pointing systems such as the "Small Instrument Pointing System (SIPS) Integration Study" final report, June 1977, NAS-5-23412 Mod 78. The results of the analysis are shown in Table 4.4-5.

Although the IPS, MPM, and POINTS pointing systems can handle all of the payloads, the POINTS is more cost effective and has more mission flexibility. It can handle c.g. mounted and end mounted payloads; and multiple payloads with different pointing requirements. Therefore, it has been selected for the STR even though the decision is close and any of the systems would suffice if a slightly lower capability was acceptable.

Table 4.4-6 shows the POINTS system capability as presently configured. Rate accuracy is based on the maximum load moving at a constant rate. The accuracy of 80 μrad/sec is a composite of gyro, bearings, and torquer noise. However, it may be advantageous to use less costly components for future fabrication. If this is the case a study of the noise effect of the components would most likely result in larger rate accuracy numbers. In an inertial mode and tracking a star in space the rate error would be less than the 80 μrad/sec.

Position accuracy is dependent on the sensor, readout and the convergence of the attitude determination system. Absolute accuracy could be as good as 50 $\widehat{\text{sec}}$ with a star crossing the sensor field-of-view.

A systems mechanization for the controls involves providing the structure and bearings, torque motors, electronics to drive the torque motors, gyro processing electronics, power supplies and digital interface equipment.

Figure 4.4-1 shows a mechanization block diagram including the partitioning of the Drive Assembly Electronics (DAE) and the Systems Interface Module (SIM). The DAE is located on the STR section near the POINTS structure while the SIM is located as a

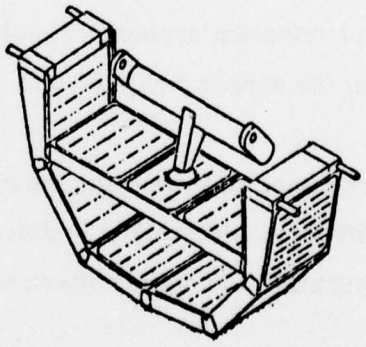
Table 4.4-5. Pointing System EPM LCC Costs (M$)

	IPS	SIPS	ASPS	MPM	POINTS
Acquisition	5.1	3.0	3.0	1.8	2.1
Operations Integration	3.0	3.0	3.0	3.0	3.0
Logistics					
NR					
AGE	0.6	0.5	0.5	0.4	0.6
ASE	0.4	0.4	0.4	0.4	0.4
Spares	2.1	1.2	1.2	0.6	0.9
Training	0.1	0.1	0.1	0.1	0.1
R					
Maintenance & Repair	0.5	0.5	0.5	0.5	0.5
Total	11.8	8.7	8.7	6.8	7.6
Cost/Flight	0.30	0.22	0.22	0.17	0.19

Table 4.4-6. Points Capabilities

SYSTEM CAPABILITIES

- AZIMUTH GIMBAL FREEDOM: $\pm 180^0$
- ELEVATION GIMBAL FREEDOM: -60^0 to $+90^0$
- TORQUE AVAILABLE: AZ = 44 FT - LBS: EL = 18 FT - LBS
- NOMINAL SLEW VELOCITY: 2 DEG/SEC (1500 LB PAYLOAD)
- NOMINAL ACCELERATION: 4 1/2 DEG/SEC^2 (1500 LB PAYLOAD)
- POSITION RESOLUTION: 20 ARC SEC (16-BIT ENCODER)
- POSITION ACCURACY: DEPENDENT UPON POINTING SENSOR USED
- RATE ACCURACY: 80 μ RAD/SEC
- RIPPLE TORQUE: 0.003 FT - LBS
- GIMBAL ASSEMBLY WEIGHT: 450 LBS
- PAYLOAD CAPABILITY: 1500 LBS
- POWER AT STALL: 70 W EL: 224W AZ



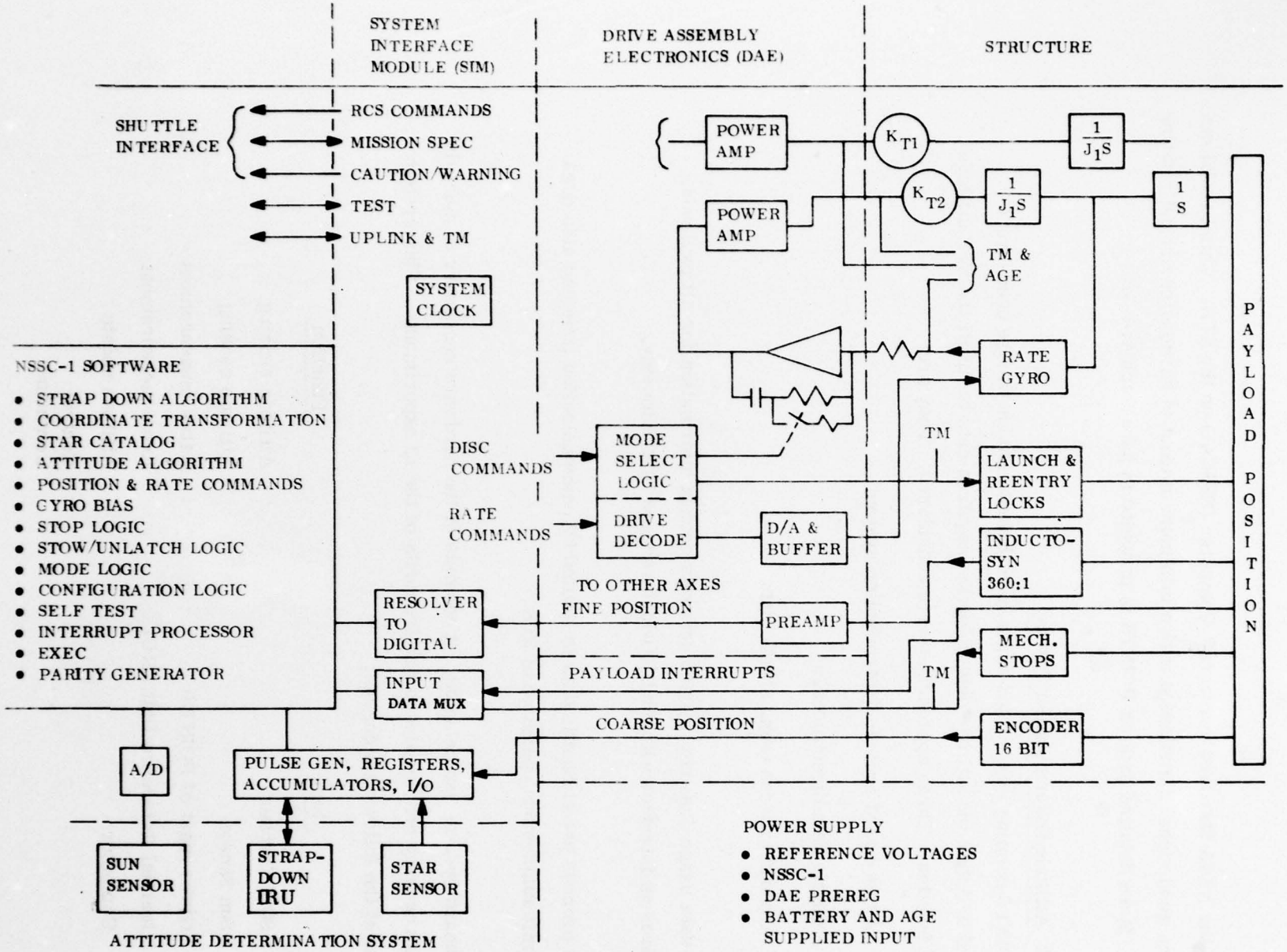


Figure 4.4-1. Mechanization Block Diagram Points

part of the NASA Standard Spacecraft Computer (NSSC-1) on the STR. This breakdown provides good signal distribution with a minimum amount of harnessing around rotating joints. More detailed data on POINTS is provided in paragraph 4.4.3.6.

4.4.2.2 Attitude Determination System

The POINTS pointing system and experiment orientation knowledge needs were analyzed to define the attitude determination requirements to fulfill the STR attitude control function. Three approaches were considered. They are:

1. An on-board attitude determination system
2. Shuttle data direct useage
3. Ground process using shuttle data

Shuttle data usage does not provide accurate enough information for all payloads. It requires an interface with Shuttle which reduces STR autonomy.

Ground process use of Shuttle data is limited by communication time and also does not permit autonomous operation of STR.

An on-board system was selected. It consists of the hardware necessary to meet the pointing accuracy and knowledge requirements of the 12 experiments. This system consists of the following elements:

Item	Function
• Star Tracker	Attitude sensing
• Sun Sensor	Attitude sensing
• Gyros (part of POINTS)	Rates measurement
• Inertial Reference Unit (IRU)	Rates measurement
• Computer	Attitude update
	Control Command

A block diagram of the attitude determination system and its relationship to POINTS and other STR subsystems is shown in Figure 4.4-1. More detailed data is provided in paragraph 4.4.3.1.

4.4.2.3 Weight and Power Summaries

The weight and power summaries for the Attitude Control System elements are given in Tables 4.4-7 and 4.4-8.

Table 4.4-7. Weight Summary

POINTS	
Structure	240
Gyros (2) & Electronics	10
Drive Assembly Electronics	35
Pre-regulator and PCM	25
Encoders and Electronics (2)	10
Torquers Elevation (2)	42
Azimuth	58
Harness Azimuth (slipring)	20
Pitch (flex)	10
	450
Attitude Determination	
Computer & SIM	18
IRU	35
Star Tracker Assembly (2)	40
Sun Sensors (2)	4
Harness	9
	106
TOTAL (WITH POINTS)	556

Table 4.4-8. Power Consumption Estimate
(At +28 Line Voltage)

Component	Power Peak (Watts)
DAE	86
Motor (1AZ)	100
(2 Elev)	56
Power Converter	83
Computer	84
SIM	15
Encoder	1.4
Encoder Power Supply	7.6
Gyro (1AZ)	62 (Full HTR)
(1 Elev)	62 (Full HTR)
Star Tracker	31
IRU	80 (Full HTR)

4.4.3 ATTITUDE CONTROL COMPONENTS

4.4.3.1 Performance of the Attitude Determination System

Since the Shuttle IMU could have a difference of 2° from the cargo bay alignment a separate attitude determination system must be provided on the STR. A strapdown IRU with fixed head star trackers mounted as shown in Drawing STR003 sheet 2 in Appendix E provide 3 axis orientation of the payload. For improved accuracy the star sensors and IRU can be put on POINTS if required. This system tracks stars of opportunity in its field of view and matches them with a star catalog to update the IRU between star encounters. Shuttle attitude could initialize the system for fast convergence of the attitude algorithm. Since convergence is not required in less than a few hours, the ephemeris information, with the orbiter in Z local vertical mode is used for initialization (eventually, GPS could be used to provide ephemeris information by adding a GPS receiver to the STR). This provides a high degree of autonomy. An uplink has been chosen as the best compromise between complete independence and Shuttle dependence. A sun sensor also provides a means of initiating faster if the Shuttle can be commanded separately. The IRU and star tracker are the NASA standard components used on MMS per GSFC-S-712-9 and 10 respectively. The

NSSC-1 computer is used to process the information. The software to perform these tasks has been resized to the 5000 words for this application using a basic 16 bit machine. This attitude determination system uses a Kalman filter to obtain estimates of gyro bias to be updated when a new star crosses the 8° field of view of the star tracker. Table 4.4-9 shows the characteristics of the sensor and its detector electronics. The inertial position is then used to estimate which star is encountered next, as the system moves under local vertical control in a particular orbit. A full star catalog could take 1808 words for 452 stars but can be cut down for a particular mission. The latitude and longitude difference between two stars encountered in a specific time is used in the Kalman to estimate gyro biases as the system converges to the accuracy less than 15 sec (2 σ).

Table 4.4-9. Star Tracker Characteristics

Null Accuracy	± 30 $\widehat{\text{sec}}$ + ± 10 sec
Null Stability	10 $\widehat{\text{sec}}$ (1 σ)
Noise Equiv Angle M=4.0 in TFOV	11 $\widehat{\text{Sec}}$ RMS
Linearity & Distortion Uncalibrated	± 3 $\widehat{\text{min}}$
Linearity & Distorltion Calibrated	10 $\widehat{\text{sec}}$ (1 σ)
1LSB	3.6 $\widehat{\text{sec}}$

TFOV: 8° in two axes

Search Period: < 10 sec time

Spatial Separation: > .6°

Motion: ≤ .3°/sec

Brightness: Selectable from +5.7 to +2.0 visual magnitude, Class GOV

Temperature: Operation between $-10° < T < +50°C$

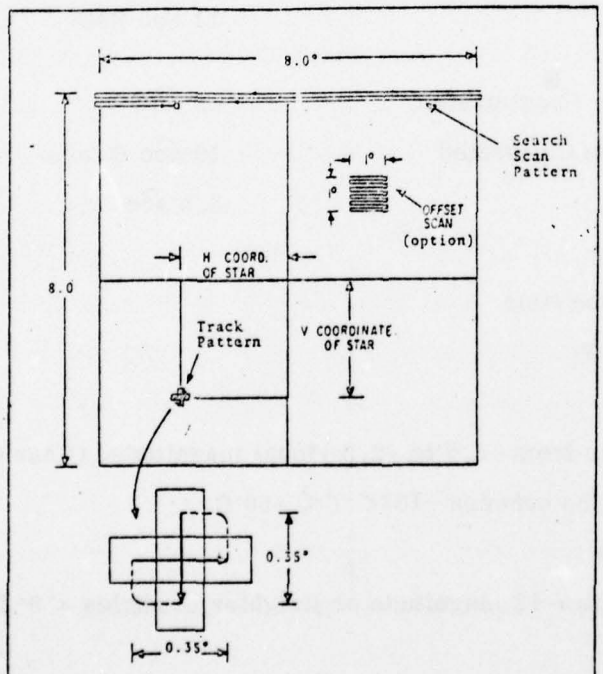
Magnetic: < .5 Gauss

Damage limit: Objects - 12 magnitude or Brighter at angles < 9° from null axis

Total uncertainty:

$$E=2\sqrt{(10)^2 + (10)^2 + (11)^2} + 10 + 1.8 \cong 50 \ \widehat{\text{sec}}\ 2\sigma$$

The tracker is an image disector and goes through a search pattern to locate a star in its field. Figure 4.4-2 shows the pattern for a scan and lock on a star. The threshold for detection (M> +6 THRU M> +3 ONLY) can be set from a software command programmed for a particular star catalog. A search of the entire field-of-view is done in 10 seconds so that a star can move .6° with the Shuttle in Z local vertical mode. The quantization for the digital readout is 12 bits plus sign resulting in a readout change of 6 bits in 0.1 seconds of update time. This is adequate for attitude determination for acquisition and tracking to within 15 $\widehat{\text{sec}}$. Figure 4.4-3 shows the convergence for a particular set of tracker angles on a particular mission. This tracker depends on the sensing of current in a magnetic deflection coil to give accurate information and has its specific accuracy for a magnetic field of less than 0.5 gauss. The location of the tracker for best alignment is as close to the POINTS dc motors as possible. For this reason magnetic shielding must be provided in the Star Tracker mounting assembly so that slew current into the torquers does not give erroneous position information. Also the digital computer program could be inhibited during a slew to prevent the magnetic



Total Field-of-View (TFOV) as viewed from inside the FHST looking out at the star field with the interface connectors facing down.

Figure 4.4-2. FHST Scan Geometry

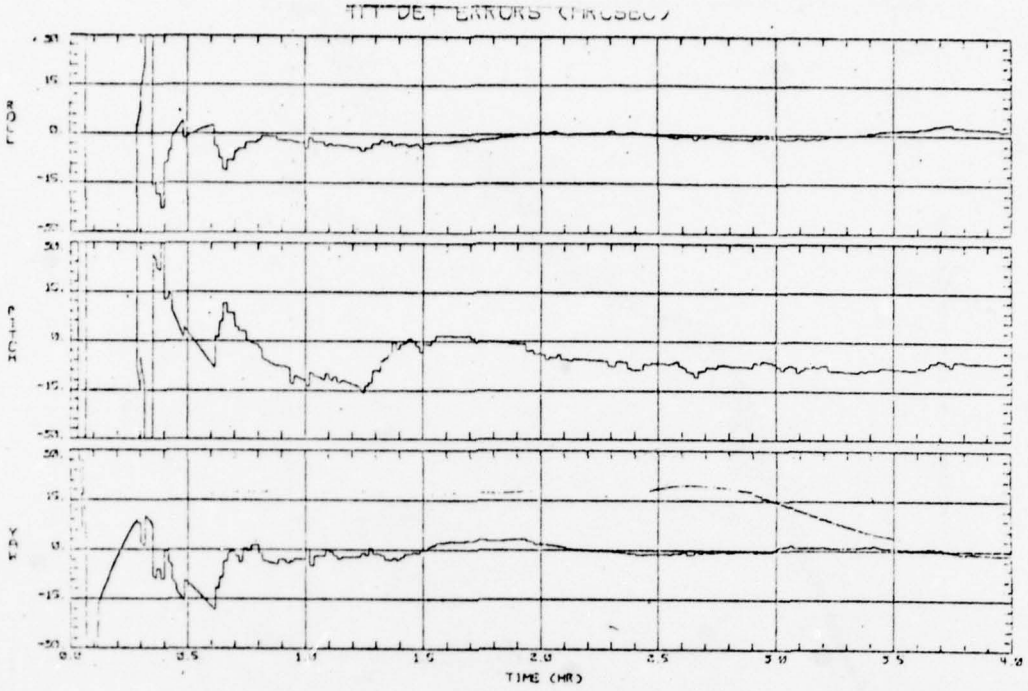


Figure 4.4-3. Simulated Attitude Errors for a Two Star Tracker System

spikes from upsetting the attitude information after it has converged. Tests will have to be run on a correctly wired system to evaluate the need for shielding of the Star Tracker. In order to prevent damage to the image disector tube a "Bright Object Detector" (BOD) must be put on the Star Tracker Housing. The BOD will activate a shutter to cover the optics of the Star Tracker in the presence of the sun, moon, Earth or targets brighter than -8 magnitude. This shutter must be activated by light before the radiation strikes the Star Tracker. However, reflections from the shuttle doors and/or payloads could also activate the shutter. A detailed optical study must be done to determine the location and number of detectors to protect the Star Tracker. The tracker should also be supplied with the self test option. The need for one or two trackers and their location must also be determined.

4.4.3.2 Sun Sensor

The attitude determination sensors are mounted on one block for best stability. The sun sensor data will be the output of two matched sensors mounted each 44° from the star sensor. Its output is shown in Figure 4.4-4. In this way it will detect when the

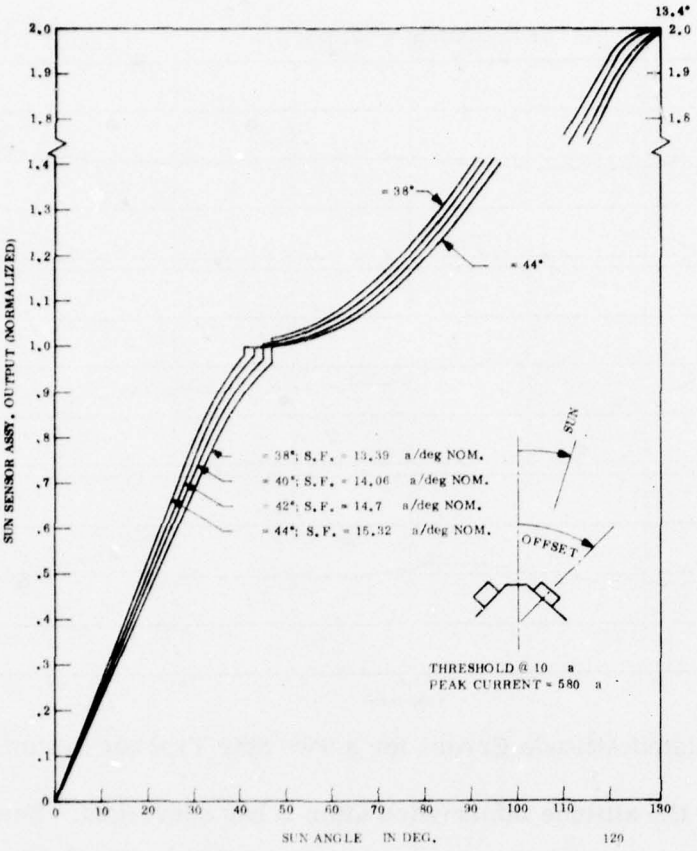


Figure 4.4-4. Sun Sensor Output

sun is approaching the star sensor field-of-view. The Star Tracker BOD will send a signal to close the star tracker shutter. With the shutter closed attitude information can no longer be processed. During this time the sun sensor will provide information until the star tracker is far enough out of the sun's radiation.

4.4.3.3 Systems Interface Module (SIM)

This assembly is the I/O for the NSSC-1 which interfaces with the other attitude control system components and POINTS. A computer I/O can either be general in nature like the STINT or it can be a special purpose design. In order to avoid redesign of many of the components in the existing POINTS system a special purpose I/O mode is used. The SIM provides the only interface to the Shuttle Systems from the POINTS. It is mounted on the NSSC-1 computer and mated with it electrically. It is located on the same panel as the star sensor and IRU for attitude determination. A power converter module will take secondary power from the Power Supply and provide isolation and EMI protection for the computer. An estimate of the sizing is shown in Figure 4.4-5.

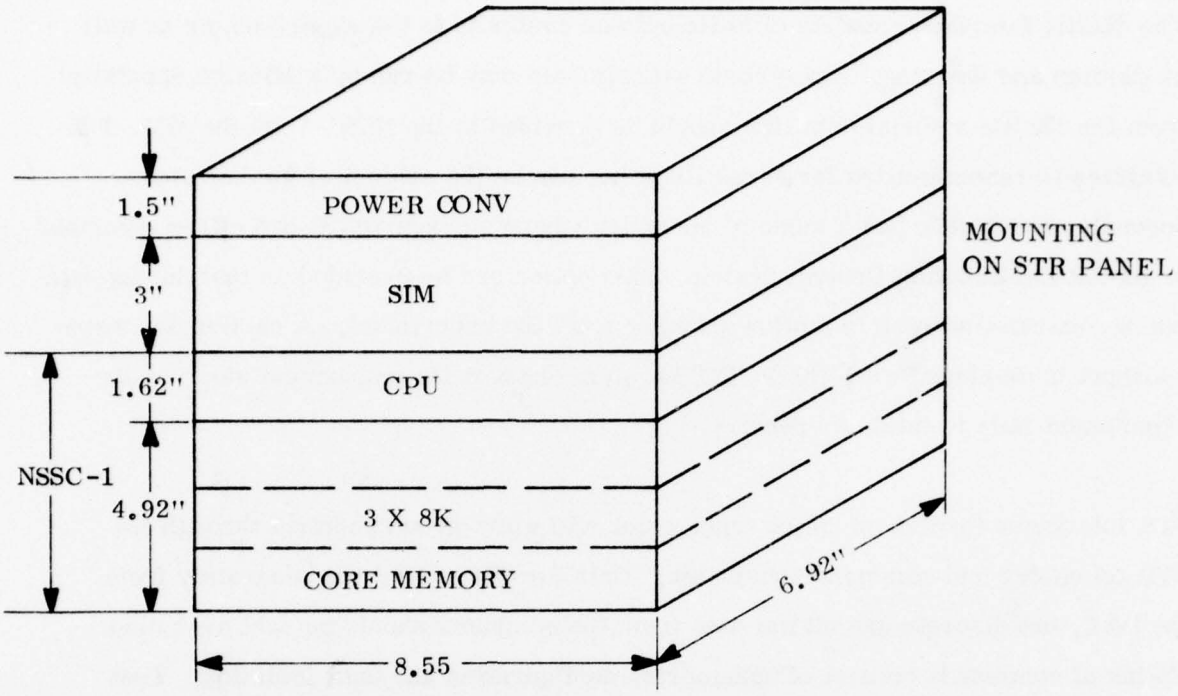


Figure 4.4-5. Computer

Functions performed by the SIM are:

1. System clock 2 MHz bi-phase.
2. Buffering and A/D conversion for the sun sensors.
3. Resolver to digital conversion for the inductosyn.
4. Pulse generation for the IRU.
5. Star sensor interrupt processor.
6. Payload interrupt processor.
7. Rate loop commands for 3-16 bit words in serial format.
8. Discrete commands for setting amplifier gains, lock release, power amplifier connections, back-up modes, etc.
9. Discrete input processing of stop engagement, latch secure, etc.
10. Encoder register for holding position information.
11. Shuttle and STR interfaces.

The Shuttle interface consists of basic options available to the experimenter as well as Caution and Warning. Since some experiments may be run by a Mission Specialist from the Shuttle a serial data link should be provided to the NSSC-1 via the SIM. This interface is recommended for versatility, but can be the subject of further study. Secondly, the Shuttle has a mode of operation where an experiment can either command or inhibit the Reaction Control System. This option can be provided so that during data taking, extraneous vehicle motion does not spoil the experiment. A caution and warning input is displayed when the POINTS system engages its mechanical stops or the experiment fails to latch for reentry.

STR interfaces involve an uplink input which will allow ground control through the STR telemetry and command subsystem. Data for TM is sent out separately from the DAE, but discrete and digital data from the computer should be sent over also. Uplink of commands consist of ephemeris, configuration and fault isolation. Test outputs from the system should go via the SIM during ground check out. Computer loading and verification, built-in-test functions, as well as an automated check out of the system can be performed via the test inputs. This would reduce turn around time and provide AGE with fault isolation information.

4.4.3.4 Power Supply

The power supply for the system will take battery power during on-orbit operation, but will be operated from AGE power during prelaunch, checkout, and integration testing.

It must convert power to secondary voltages for distribution to SIM, NSSC-1, DAE and sensors. It must also provide the boost regulation for the DAE power amplifiers to be run at +34 vdc, ac references must also be generated. For example: the inductosyn requires a 10 kHz excitation.

4.4.3.5 POINTS Pointing System

4.4.3.5.1 Structure

POINTS installed in the Standard Test Rack is shown in Figure 4.4-6. The heart of the POINTS system is a gimbal assembly with two (2) axes of rotation. It can hold a payload mounted between two pivots and bearings, seven (7) ft. apart. The pivots in elevation can move one payload of 1500 lbs and two at 750 lbs independently. This feature is possible because of a dual drive system built into the upper assembly. The lower assembly is a precision drive about the azimuth axis (Shuttle Z axis). Approximate dimensions for the structure is shown in Figure 4.4-7.

The existing structure is made of Beryllium for light weight strength. It can be redesigned as an aluminum assembly at lower cost of fabrication, in production quantities. To accommodate launch and landing loads, mechanical stops are provided. These are to prevent failures in the system that could impact flight safety. These stops are relocatable and can be changed as part of the particular payload integration. A combination (Launch lock and stowing) locking mechanism is implemented on the structure for STR. The present system is designed for just one specific payload and is commanded from software. For some shuttle payloads (end mounted, long structure like HIRISE) a locking mechanism may have to be located on a structural extension.

This POINTS structure along with its bearings, contain the dc torquer motors and feedback elements to precisely control the motion of the payloads, including tracking specific points on the ground.

4.4.3.5.2 Control Mechanization

Payload control is achieved through a software controlled inertial rate loop within a position loop. The angular command for position is based on precise inertial position as derived from the attitude determination system.

Figure 4.4-8 shows the control system block diagram. For each axis the control mechanization is the same. In single payload cases the redundant drive motor is connected. In dual payload cases, the redundant drive and gyro are used to command

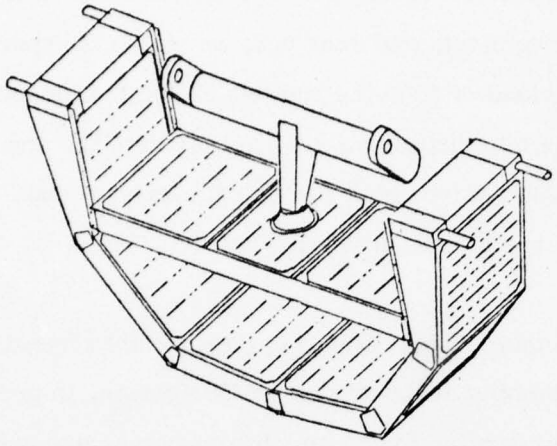


Figure 4.4-6. POINTS Installed on Bridge in STR

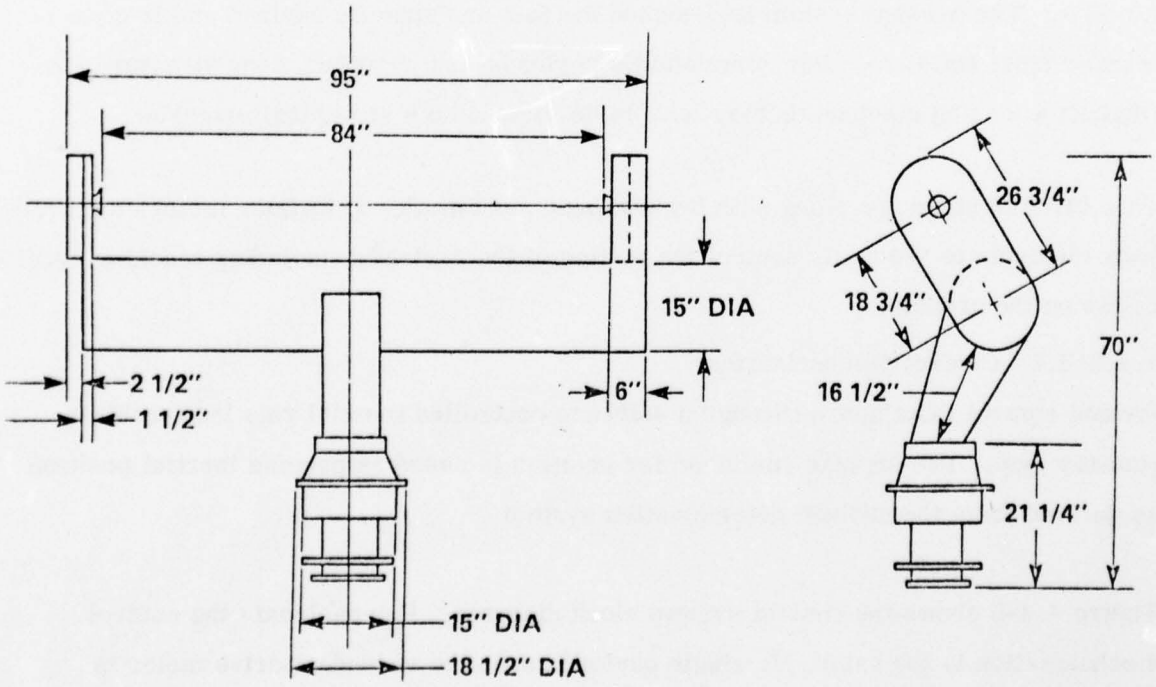


Figure 4.4-7. POINTS (Upper Assy)

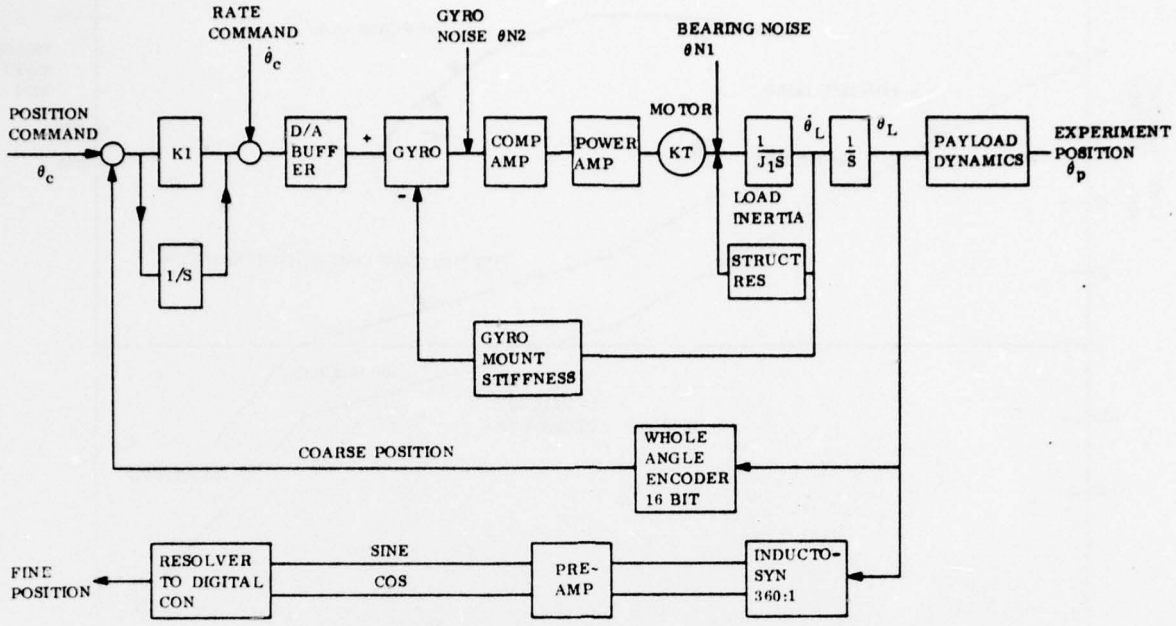


Figure 4.4-8. Control System Block Diagram

a separate payload in elevation. A rate command from a digital computer is converted electronically to drive the gyro at a precise inertial rate. The gyro output is fed into a compensation amplifier and current power amplifier to drive the gimbal motors. The load moves at the commanded rate minus the friction effect thus achieving precision control. A bode plot of the loop as mechanized is shown in Figure 4.4-9. The loop compensation is designed for a crossover at 15 rad/sec by providing the proper gain for the maximum inertia and a lag at low frequencies. However the POINTS system must be mechanized to provide control for a wide range of payloads and therefore the load inertia could be a lot less.

Figure 4.4-9 also shows what happens to the frequency response curve if a small payload (1/10 the maximum) is put on the gimbal. Under these circumstances the system would be unstable. A range of compensation is provided and commanded by software to prevent system instability.

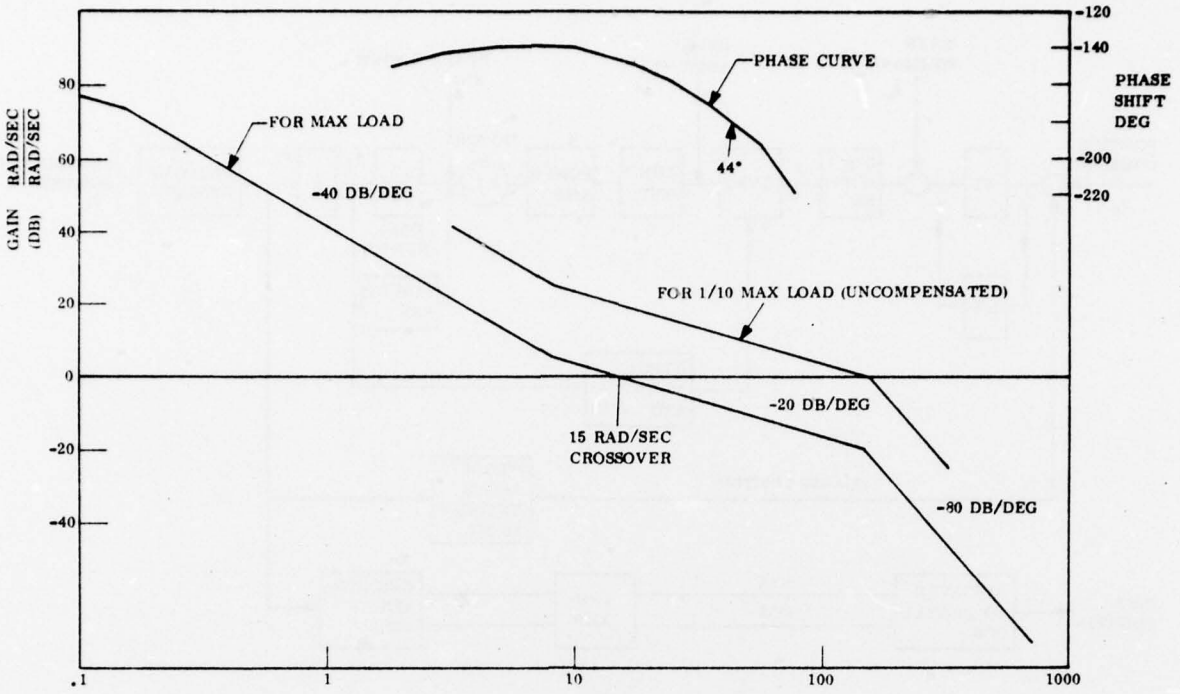


Figure 4.4-9. Frequency Response of POINTS Drive System

4.4.3.5.3 Torque Disturbance

In the analysis of the systems jitter the POINTS mechanization was used with maximum load. For payloads that must track an Earth target smooth rate is important. However, the smoothness of the drive will be less for smaller payloads. In general, the smoothness is a composite of all noise sources and is shown in Figure 4.4-8 as a gyro noise response input. To illustrate the reduction in smoothness due to bearing noise, (ENI) the torque disturbance plot of Figure 4.4-10 is drawn. The lowest curve shows the response for the maximum load system. A ripple torque of 1 ft-lb at 4 rad/sec will result in a payload jitter of -70 db or 1/3 at 10^{-3} rad/sec. If the payload is 1/3 the maximum inertia, the error goes up by 10 db or 3 times. The resulting jitter is still a very low level of 1×10^{-3} rad/sec.

4.4.3.5.4 Position Control

The digital computer provides control functions to implement a gimbal position hold loop. As shown in Figure 4.4-8 a position command (Q_ϕ) is processed in the computer

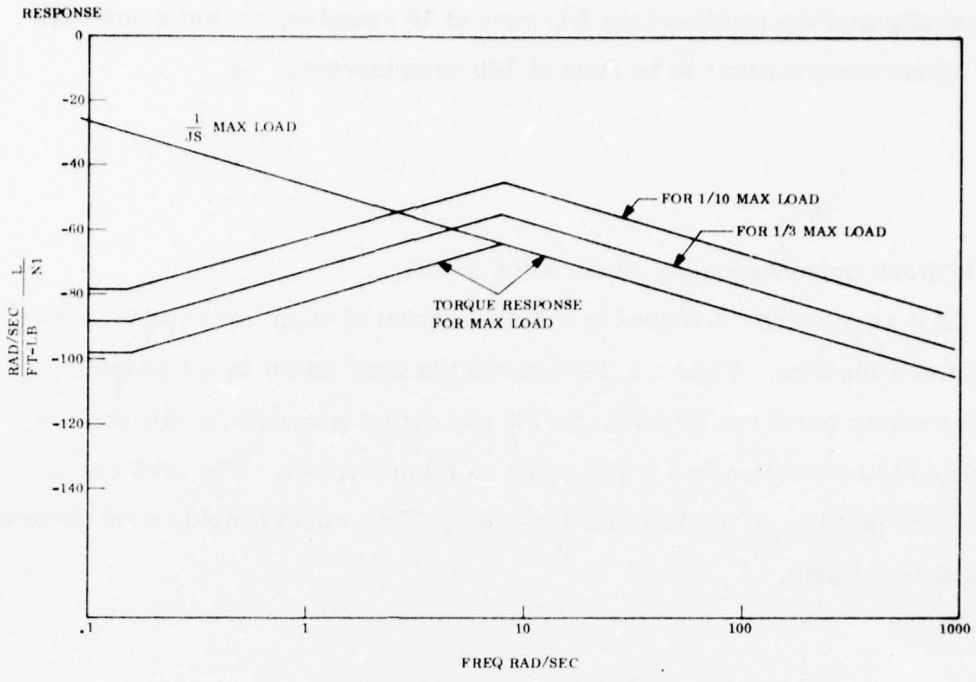


Figure 4.4-10. Torque Response for Various Load Inertia of "POINTS" (Compensated for 15 rad/sec loop crossover)

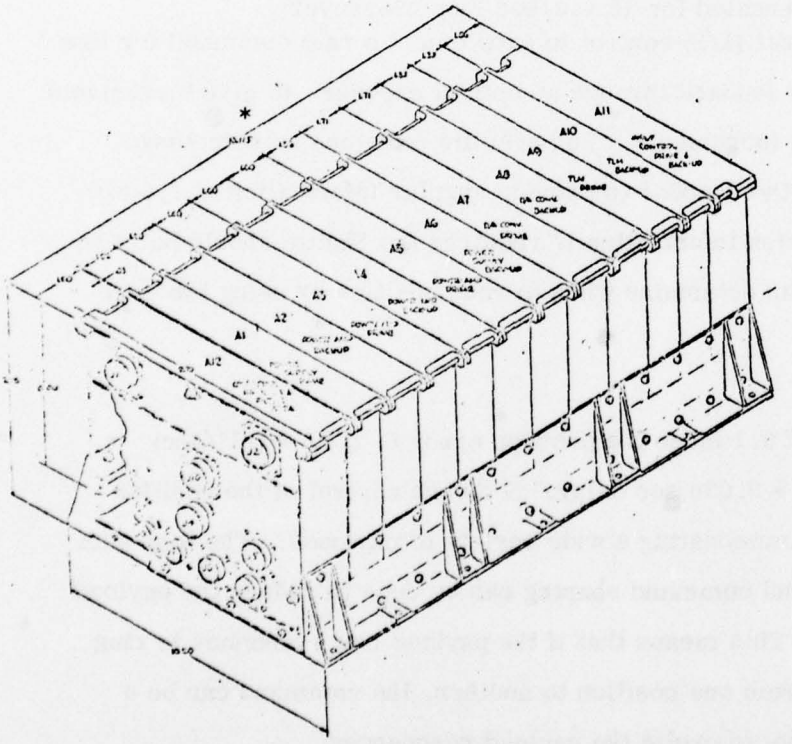
with proportional (K) plus integral (1/S) control to supplement a rate command for fine pointing. The output position is fedback through an optical encoder to give increments of gimbal position in 20 arc sec increments. For specific missions an inductosyn readout is provided to augment the encoder to provide angular information at 1.0 arc sec resolution. When fine position information is required the Shuttle should be in an inertial mode and the payload can determine when to read position by using the computers interrupt capability.

Using the interrupt capability of 0.1 msec the position error is $\Delta\theta_L$ = (11°/sec) (3600 sec/deg) (0.1×10^{-3} sec) = 0.036 sec delay. Software control of the position loop provides versatility in accommodating a wide variety of payloads. The loop gain and bandwidth can be selected and command shaping can be done to reduce the payload response to step acceleration. This means that if the payload has a tendency to ring dynamically when accelerated from one position to another, the command can be a slowly changing function so as not to excite the payload resonances.

As presently configured the position loop I/O runs at 10 samples/sec but a software extrapolator allows computations to be done at 100 samples/sec.

4.4.3.5.5 POINTS Drive Assembly Electronics (DAE)

The POINTS DAE as presently designed is a fault tolerant system. No single failure can cause loss of a function. Figure 4.4-11 shows the card layout in the assembly. Because more modern parts can be used the TM and digital processors will probably be accommodated in less volume as is indicated on Figure 4.4-11. The DAE can be located on the STR panel near the POINTS assembly. This would provide good structural support and a thermal path.



*Dimensions Depend on Reconfiguration (EST 14.5")

Figure 4.4-11. Drive Assy Electronics

The functions the DAE must perform are shown in Figure 4.4-1 and are:

1. TM processing of analog data
2. AGE outputs for test
3. 4-Power amplifiers to drive 4 dc motors

   1.5a maximum in each elevation axis
   5.2a maximum in each azimuth axis

4. Provide loop compensation for each axis
5. Compensation changeable for different loads
6. Change compensation via software discretes
7. Provide torquing signals to drive each of 3 gyros independently from a D/A converter.
8. Provide amplification for sine and cosine windings of 2 Inductosyns.
9. Decode latch commands, stow commands and Back-up mode configuration changes.
10. Decode the logic from input addresses to be sent from the SIM in serial form. One word for discrete commands, and 3 for gyro commands.

4.4.3.5.6 Software

The functions to be performed by the POINTS system are under computer control. Software must be designed for these functions, but must be kept on a modular basis to satisfy a variety of missions. The functions listed in Figure 4.4-1 receive inputs from hardware and must be kept resident at all times. The mission dependent variables will be in alterable memory locations and merely changed for each mission. The STR Mission Manager prepares a data tape to change each mission data and verify the contents at final checkout of the STR with POINTS. Variables to be changed would include:

1. Gain settings to accommodate particular loads
2. Configuration - 1 or 2 elevation drives

3. Stop positions

4. Stow position for reentry

5. Control commands for the particular mission

6. Star/sun catalog changes

Table 4.4-10 is an estimate of the 16 bit words required to do the tasks listed in Figure 4.4-1.

Table 4.4-10. Software Sizing

Task	Word Size in NSSC-1
• Strapdown Algorithm	2000
• Coordinate transformation	500
• Star Catalog	1808
• Attitude Algorithm	5000
• Position & Rate Commands	500
• Gyro Bias	100
• Stop logic	200
• Stow/Unlatch Logic	100
• Mode Logic	200
• Configuration Logic	100
• Self test	2000
• Interrupt processor	500
• Executive	2000
• Parity Generator	100
	15108
Spare x 1.5 (size for 3 memories) at 8K per memory)	24000

4.5 STRUCTURE SUBSYSTEM

The STR must provide a structure which meets mounting and strength requirements of the STS and the twelve identified experiments.

4.5.1 STRUCTURE REQUIREMENTS AND DESIGN CRITERIA

The Structure Subsystem must accommodate a wide variety of experiments using modular subsystems that may be increased or decreased in size and performance capability to meet the needs of individual experiments.

It must also permit integration installation of structure and subsystems into the shuttle bay cabin quickly with minimum time and effort.

Additional requirements include compatibility with the NASA five point attachment and with installation around the Spacelab tunnel. The structure may not yield at crash ultimate loads and must be capable of supporting up to 5000 pounds, including both STR and payloads.

4.5.1.1 Design Criteria

The requirements presented in the following sections are based on the latest available STS information obtained from current documentation. These requirements provide the design basis of the STR primary structure, secondary structure, and its components. It will be noted that extensive studies are continuing to better define the Space Shuttle environments and loads. Although the requirements are current, they must be treated as preliminary and revised as additional analysis and experimental data become available.

A review of the available documentation relating to shuttle accommodations and shuttle/payload coupled dynamic analyses (e.g., SAMSO TR-76-212 "DOD Shuttle Integration Support Study", Boeing Report "IUS System Design Review", MDCG6626 "PAM User's Requirements Document") show the load factors could be higher than

those used in this criteria. These load factors are influenced by the payload geometry and dynamic coupling with the shuttle in much the same manner that current spacecraft/launch vehicle coupling influences spacecraft design loads. Although various maneuver and boost conditions have been examined, results to date have indicated that lift-off and landing are probably the most critical for payload design. For these conditions, the dynamic loads can be greatly magnified by the method of supporting the payload within the cargo bay. Maximum loads during lift-off and landing are significantly reduced by supporting the payload according to basic guidelines followed in the STR design. These guidelines include: (1) minimize lateral asymmetries, (2) provide balanced stiffness in fore and aft supports, and (3) locate fore and aft supports near the radius of gyration of the cargo assembly.

The STR was designed with very high margins of safety. Thus, the capability to withstand higher load factors was incorporated into the design by utilizing the following design criteria.

1. No yield at crash ultimate loads.
2. No more than 1/2 yield at 2σ flight limit loads.

These criteria eliminate fatigue failures for normal operation since the endurance limit of the materials is approximately 1/2 yield strength.

The STR structure possesses sufficient strength and rigidity to provide the necessary structural support and housing for the various payloads and to survive the required critical loading conditions that will exist during all phases of ground test and boost flight. Methods of controlling ground-handling transportation, and storage environments will be used to avoid developing critical loadings. The structure is designed to achieve minimum weight commensurate with requirements.

The details of the design criteria and requirements can be found in Appendix C.

4.5.2 STRUCTURE DESIGN SUMMARY

The Standard Test Rack structure is constructed of 9 inch deep formed channels, 4 x 4 angles and 0.1 inch thick doublers, caps, gussets and plates of 6061 T6 aluminum that are mechanically fastened to provide a structure that can be constructed in many configurations to support a wide range of experiments and equipment (see drawings STR 001 to STR 011 in Appendix E). Seven different constructions of the basic STR model are shown in Figure 4.5-1 and all provide 6 equipment bays, of equal size, in the main structure with additional equipment bay capacity provided by the addition of a 9 inch deep, 47 inch wide beam that spans the 141.2 inch open end of the structure and which can be adjusted vertically along a 45.5 inch span of attaching holes provided at the open end sections of the STR or by addition of a strut supported, 9 inch deep, 47 inch wide shelves attached at the open end of the STR structure. Typically the shelves could be 42 x 47, 9 inch deep bays commensurate with the space required for the components to be mounted.

The seven STR configurations shown in Figure 4.5-1 are:

1. Full STR with upper bridge that utilizes maximum space available around spacelab tunnel or other locations and permits total STR usage.
2. Full STR without bridge can be mounted under Spacelab tunnel or other locations.
3. Full STR with dual shelves.
4. Full STR with a single shelf.
5. Full STR with POINTS mounted on bridge.
6. Full STR with POINTS mounted on bridge (side mounting).
7. Full STR with POINTS mounted to equipment panel.

The seven STR construction options shown are all constrained to outboard dimensions within the 87.8 inch radius shuttle cargo interface envelope and to inboard dimensions that assure ample clearance with the 47.4 inch diameter, air lock tunnel interface

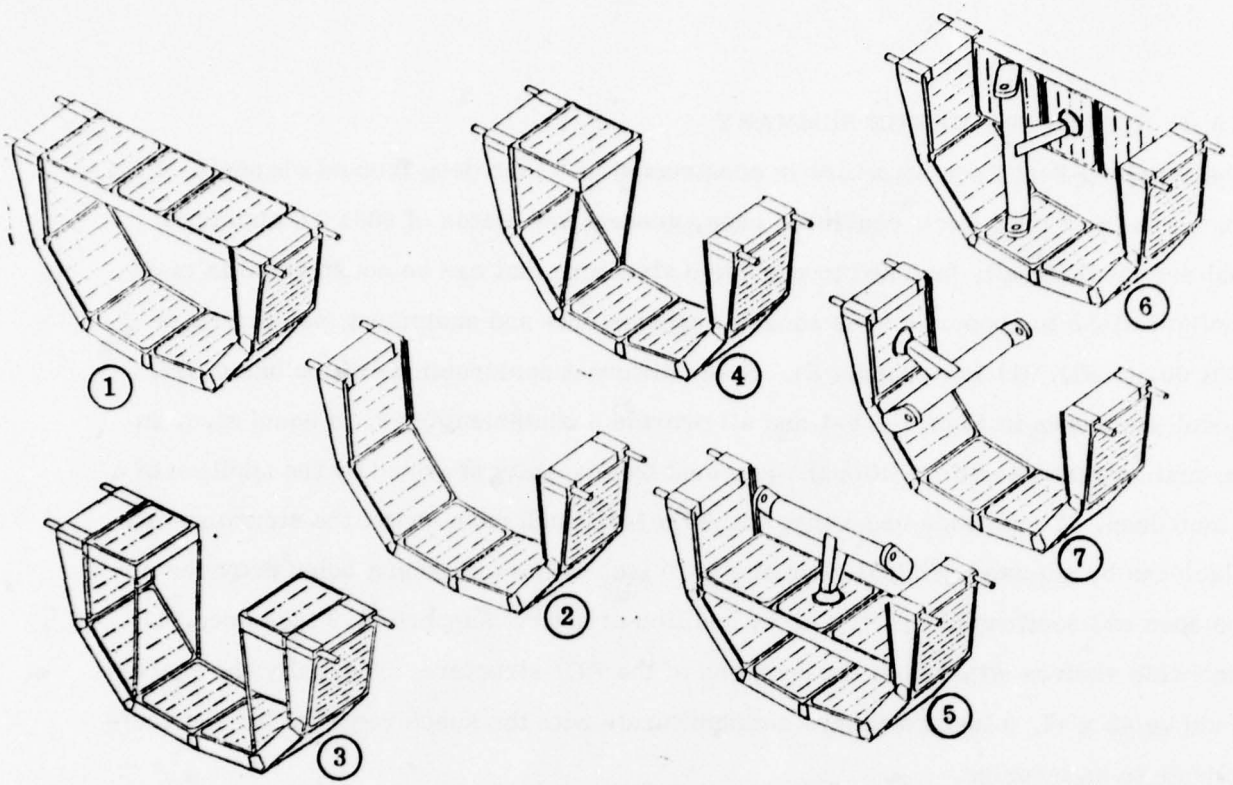


Figure 4.5-1. Construction Variations with Basic STR Structure

requirement. The STR structures incorporate the 4 trunnions at the top and a keel fitting at the bottom for installation in the shuttle. See Drawings STR 001 to STR 003 in Appendix E for construction details.

The six structural bays or equipment modules at the midsection of the STR are identically constructed, are interchangeable and are basically 9 inch deep, 47 inch wide sections with 43 inch inside chord length and 48.4 inch outside chord length equipment compartments. Hat section stiffeners are added to the inside of the closure panels which are basically 42 x 47 inch surfaces that are designed to be fastened to the structure of the STR and to provide the structural support, positioning and temperature control for the components and experiments which are packaged within each of the 6 main equipment modules of the STR. Each of these STR equipment compartments provides a little more than 10 ft^3 of useable volume and therefore avionic or electronic components that customarily have packaging densities of 20 to 25 lbs/ft^3 can be accommodated with equipment weights of 200 to 250 pounds per module. Heavier weights (approximately 500 pounds) of denser components such as batteries

and power conditioning components can be packaged in each compartment and up to one hundred pounds of the less dense equipment (where more space is usually needed for deployments, calibration targets, etc.) could be considered for packaging/planning for each module of the STR structure. In the final STR flight configuration it is expected that weights and balance control mass will be required to achieve proper weight distribution, center of gravity and inertial properties values.

The geometry of the structure has been selected to allow for the STR to be installed in the Shuttle with the air lock tunnel in place and with the Shuttle in either the horizontal or the vertical position. The STR can be initially positioned with the STR open end over the tunnel, then rotated about the tunnel, using ground support equipment, constrained within an outside radius of 90 inches until it is positioned with the STR surrounding the tunnel and the open end positioned up in the Shuttle cargo bag (see Figure 4.5-2). In a contingency situation the STR can be removed by reversing the operations for an open-end structural model of the STR. The bridge beam could also be added or removed when conducting these procedures with a closed-end STR model by using ground support equipment attached to the beam and the basic open end STR structure. The bridge would be released or attached by removing or installing the bolts at the open end structure fittings.

Drawings (STR 004 to STR-011 in Appendix E) show the relative size, position and arrangement of 12 representative experiments using different construction models of the basic STR structure. A weight estimate and list of the main elements of the STR structure are given in Table 4.5-1. The 1480 pound estimate is for the STR construction model incorporating the bridge beam and a weight estimate of the basic, open-end structure is essentially 375 pounds less or 1105 lbs. Weight estimates of typical support equipment installed in the STR structure are included in Table 4.5-1 for power regulation and distribution equipment for 300 pounds of Lithium batteries, for telemetry, command and data handling, for attitude control and for thermal control components. For example, the 300 pounds of lithium batteries for 37.8

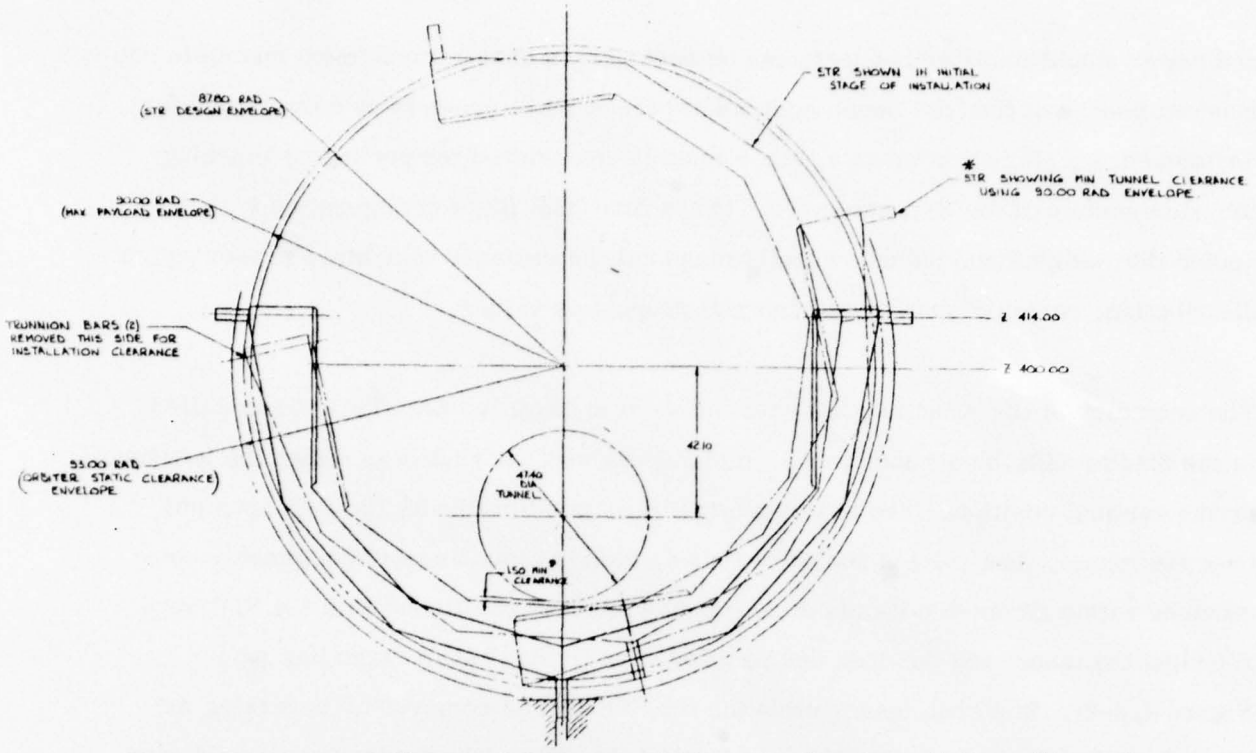


Figure 4.5 2. Installation of STR around Tunnel (after installation of tunnel)

KWH of electric power is easily packaged into one compartment to a density of 30 lbs/ft^3 and the telemetry, command and data handling equipment into one compartment at a packaging density of 10 lbs/ft^3 as shown in Drawing STR 003 in Appendix E.

4.5.3 STRUCTURAL ANALYSIS

The main criteria for designing the STR structure is that there be no yield at STS crash ultimate loads and that the stress levels be less than 1/2 yield strength at 2σ flight limit loads. This is a conservative design approach and additionally eliminates fatigue failure considerations under normal operations since the material endurance limit is essentially one half the yield strength value.

Strength analyses of the preliminary STR structural design were made of both the STR construction model that includes the bridge beam and for the basic open end STR model when it is being installed in the Shuttle (see Figure 4.5-3) with the Shuttle air lock tunnel in place and considering the Shuttle in the horizontal and in

Table 4.5-1. STR Structure Weight Breakdown

Structure		1480
Trunnions (5)	225	
Frame, Strong Back (2)	250	
Closure Panels (6)	150	
Equipment Panels (6)	330	
Frame, Bridge	130	
Closure Panels (3)	75	
Equipment Panels (3)	170	
Hardware, Misc.	150	
Shelf	(130)	
Power		130
Batteries	4.8 kwh/string (300#)	
Harness	100	
Power Regulator	20	
Power Controller	10	
Communications		99
Antenna (2)	0.4	
Hybrid (Ant.)	0.04	
Diplexer	1.5	
Switch	0.38	
Transmitter (2)	6.0	
Telemetry Unit (Red.)	13.4	
Recorder (2)	20.0	
Hybrid (Rcvr)	0.04	
Receiver (2)	6.0	
Decrypter (2)	8.8	
Command Decoder (red.)	7.6	
Antenna	0.2	
Switch	0.38	
Transmitter (2)	6.0	
Encryptor (2)	8.6	
Recorder (2)	20.0	
Attitude Control		106
Sun Sensor (2)	4	
IRU	35	
Star Tracker (2)	40	
NSSC-1 computer	18	
Harness	9	
Thermal Control		220
Heat Pipes	135	
Blankets	65	
Heat Exchanger (15 KWH)	(444)	
Misc. Attach.	20	

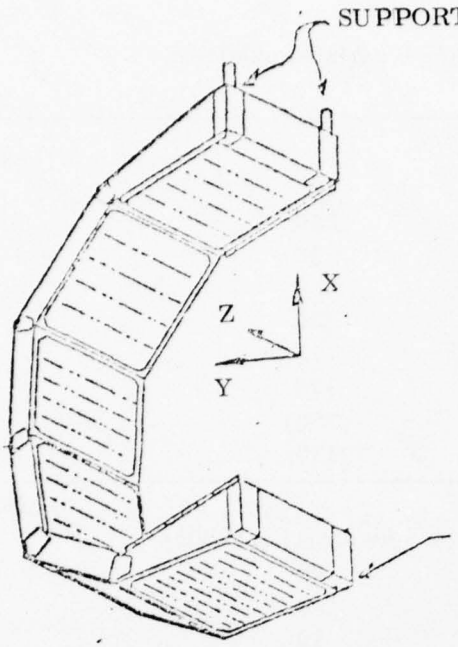


Figure 4.5-3a. Rack Free End Displacements for X-Direction Loading

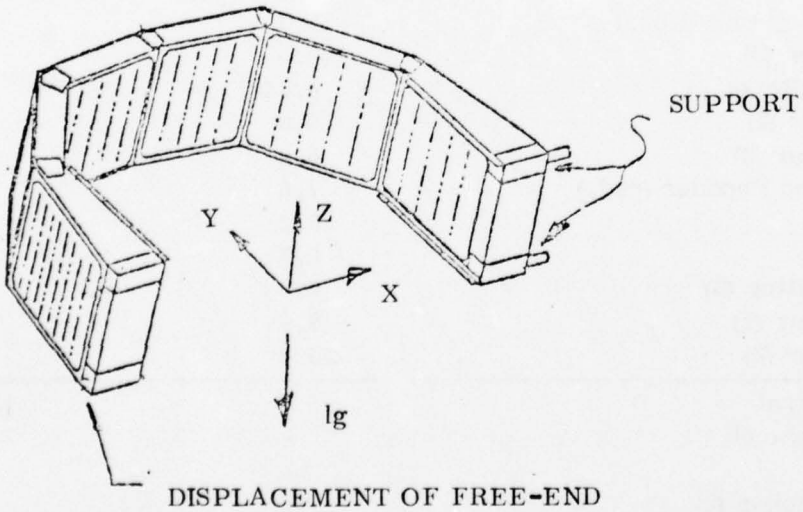


Figure 4.5-3b. Rack Free End Displacements for Z-Direction Loading

the vertical position during the installation. Both the bridged and the open type design models were analyzed for the load cases given in Table 4.5-2 and the analysis assumed that the total structure with bridge beams weights 1719 empty (conservative) and the equipment supported by the STR structure equal to 3671 pounds in a total weight of 5390 pounds. The minimum margins of safety for the 12 load cases studied of the 2 construction models are summarized in Table 4.5-3, and as expected, the conservative design approach taken results in high margins of safety. The deflections analysis during installation of the STR (Figure 4.5-3) show that only small clearances are required to prevent contact with the air lock tunnel.

The analysis of the structural strength and deflections under the critical loading conditions are given in the following sections, including the finite element models used and the calculations made for performance margin determinations.

Table 4.5-2. NASTRAN Load Cases

Load Case	Direction	Load Factor (g)	
1	X	1.	Structure Only
2	Y	1.	
3	Z	1.	
4	X	1.	
5	Y	1.	
6	Z	1.	
7	X	1.5	
8	Y	4.5	
9	Y	-4.5	
10	Z	4.5	
11	X	1.0	
12	Z	1.0	

Table 4.5-3. Standard Test Rack Minimum Margins of Safety

Item	Element Number	Load Case	Material	Failure Mode	Margin of Safety
Rack Plate t = .10 in (Inside)	2	8	6061-T6 Aluminum	Combined Comp & Shear Buckling	+5.26
Rack Plate t = .10 in (Outside)	9	9	6061-T6 Aluminum	Combined Comp & Shear Buckling	+3.81
Bridge Plate t = .10 in	86	10	6061-T6 Aluminum	Combined Comp & Shear Buckling	+8.91
Keel Fitting	54	7	6061-T6 Aluminum	Bending	+1.56
Trunion Fitting	48	10	6061-T6 Aluminum	Bending	+1.28
Longitudinal I-Beam	102	10	6061-T6 Aluminum	Crippling	+1.65
9 in. Deep Channel	94	8	6061-T6 Aluminum	Crippling	+1.55
Tapered Channel	23	10	6061-T6 Aluminum	Crippling	+2.42
17 in. Deep Channel	39	10	6061-T6 Aluminum	Crippling	+3.89

4.5.3.1 Analysis Approach to Structural Verification

The STR structure has been analyzed using standard stress analysis techniques. The analysis covered strength, instability, buckling, and appropriate elastic and inelastic theory. Combined effects were analyzed and margin of safety were based on the stress ratio:

$$R = \frac{\text{Applied Stress}}{\text{Allowable Stress}}$$

$$\sum_{\eta}^{\tau} R \leq 1.0$$

where

$$\Sigma R_{\eta}^{\tau} = R_1^n + R_2^o + R_3^p + \ldots$$

and

$$M.S. = \frac{1}{\Sigma R} - 1.$$

For purposes of determining design margins of safety, material properties were obtained from MIL-HDBK-5 and MIL-HDBK-17 and the margins of safety determined were based on "A-values" of MIL-HDBK-5 for single load path structures and on "B-values" for multiple load path structures.

4.5.3.2 Finite Element Model

The UNISTRUC computer program was utilized to develop the finite element model of the STR structure which is compatible with release number 43 of the CDC-MacNeil-Schwendler version of NASTRAN. The model consisted of 77 grid points with 455 degrees of freedom. Figure 4.5-4 shows the total model optimized grid. In modeling the structure 98 beams and 22 plates were used. The restraints put on the model represent the standard shuttle 5-point attachment method. The keel fitting is restrained in the lateral direction (X-direction) only and the trunnion fittings are restrained in the vertical and/or longitudinal directions (Y and Z-directions).

The finite element model was modified to determine the effects on the structure during installation around the STS airlock tunnel. The bridge structure was removed to provide an open end structure with the structure supported at grid points 1, 10, 19, and 28 to simulate the use of a ground handling fixture in installation of the STR.

4.5.3.3 Load Cases

The two STR structural arrangements were subjected to the load cases shown in Table 4.5-2 using the NASTRAN computer program using a total weight of 5390 lbs.

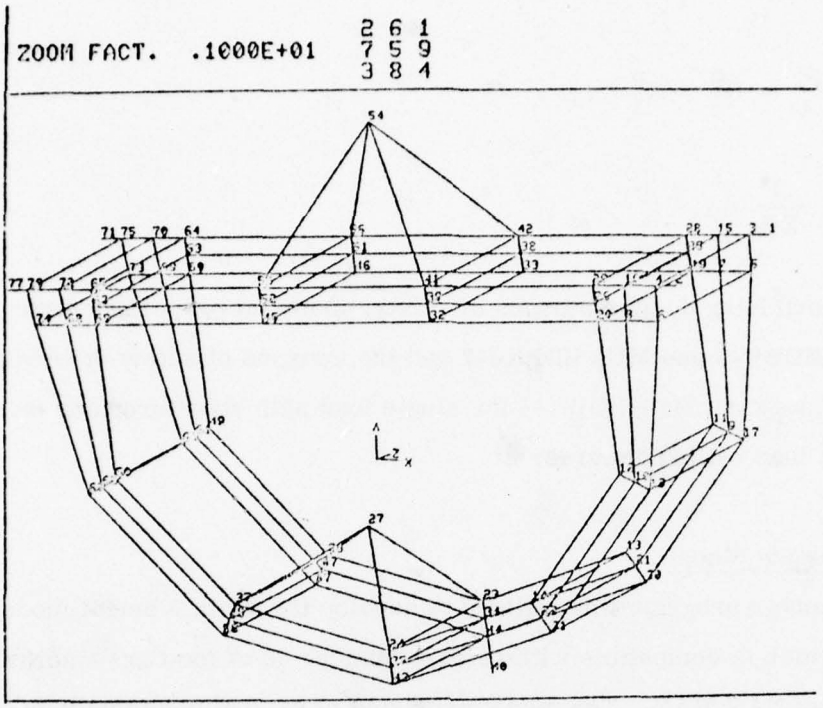


Figure 4.5-4. Standard Test Rack Optimized Grid Points

4.5.3.4 Structural Analysis Results

The finite element model was used with NASTRAN to determine the STR displacements and internal load distributions. Using these data the natural frequencies and margins of safety were calculated for the STR.

4.5.3.4.1 Natural Frequency

Using the displacements from the three unit load cases (4, 5, and 6), the natural frequency of the loaded STR structure for each of the principal directions was determined. These frequency results are shown in Table 4.5-4 and all values meet the imposed minimum requirements of 6.5 Hz fundamental.

4.5.3.4.2 Stress Analysis

Based on the NASTRAN computer results a preliminary stress analysis of the various STR structural elements has been performed. Table 4.5-3 presents the minimum margins of safety for the structure and shows that all margins are greater

Table 4.5-4. Standard Test Rack Natural Frequencies

Direction	Frequency* (Hz)
X	6.93
Y	8.52
Z	16.06
$^{*}\text{Frequency} = \frac{3.13}{\left[\delta_{\text{static}}\right]^{1/2}}$	

than 1.28. Additionally, Figures 4.5-3a and 4.5-3b present the rack free end displacements for the installation simulation. The two conditions analyzed represent the cases where STR installation is required when the shuttle is in a horizontal or vertical position. Figure 4.5-3a is for horizontal and 4.5-3b for a vertical shuttle. The critical condition occurs for installation with a horizontal shuttle. Based on the geometry of the tunnel and the STS structure a displacement of one inch is allowable and as can be seen that the STR displacements are within this allowable. No stress problems were found to occur for this configuration.

4.5.3.5 Strongback Configuration

In addition to the strongback/bridge configuration of the STR, a half strongback configuration was examined. This consisted of the configuration shown in Figure 4.5-5.

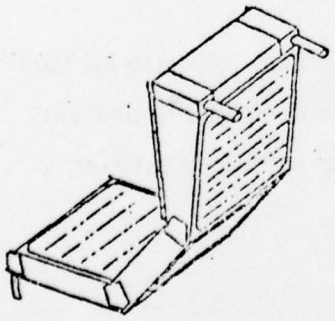


Figure 4.5-5. Half Strongback Configuration

Several support configurations were also examined. The baseline support system (2 trunions and 1 keel fitting) did not provide a stable system. To achieve a stable configuration it was necessary to have 2 keel fittings with vertical load carrying capability or both trunions must act as

fixed supports. With these restrictions it was decided that the half strongback configuration was not a viable configuration and was dropped from further investigation.

The details of the stress analysis can be found in Appendix C.

4.5.3.6 Jettison Requirements

Inherent in the safety of the STS is the ability to return from orbit with the bay doors closed, the payloads mounted securely and less than 32,000 pounds of payload remaining. Deployable systems must have a backup to insure that failure does not preclude safe return from orbit. One way to achieve this is to have an astronaut go on an EVA to either correct the malfunction or cut the assembly loose. EVA was assumed applicable as a backup only. (See Figure 4.5-6). Other approaches involve providing jettisoning capability for all items that deploy beyond the cargo envelope and provide safety stowage position for deployable items.

Several methods were examined for jettisoning failed or otherwise unwanted equipment for the down leg part of the mission. These include:

1. Marmon clamp
2. Pin pullers
3. Separating Nuts
4. Cable Cutters
5. In Flight Disconnect

} for electrical wiring

The marmon clamp is ideal for payloads with a circular interface. Tension springs to pull the band down away from the separation ring plane are desirable to prevent tipoff rotational rates. Pin pullers have been used very successfully on programs

- Jettison Capability Required
  - All items that deploy beyond cargo envelope
  - Safety stowage position for deployable items
- EVA is Backup only

Figure 4.5-6. Jettison Requirements

such as 206, BSE, Nimbus, Landsat, and others (as have Marmon clamps) and can be used with confidence where applicable. However, the first choice is separating nuts with redundant squibs in each nut. This is probably the most universally adaptably mechanism and the most cost effective. A series of separating nuts in 1/4", 3/8, 1/2, 5/8, and 3/4" bolt sizes can be made which all use the same squibs for power. They can be resettable which permits test and or operation with cold high pressure gas if desirable. They can also be conventional design or low shock design as the design dictates. For a particular configuration, the appropriate size or nut is chosen for that installation.

As an alternate or backup to the separating nuts, a band clamp assembly may be used. For the bridge, pin pullers are a possibility depending on the configuration and loads.

In summary, separating nuts provide the best combination of size to weight carrying capability and low cost for a multiplicity of applications.

The choice between cable cutters and in flight disconnect will be based on the size of the cable. In flight disconnects are used with large cables and cable cutters are applicable to small diameter cables.

4.6 TEST REQUIREMENTS

4.6.1 STR TEST FLOW

The baseline STR Test Flow Plan for development, qualification, protoflight and flight hardware is shown in Figure 4.6-1.

This test flow places emphasis on component level testing to minimize the impact of STR configuration changes to accommodate different experiments.

Testing of STR flight units is limited to electrical/mechanical integration, functional performance measurements and external interface verifications. When STR testing is complete, the STR will be ready for experiment integration and test.

Protoflight STR hardware will be subjected to additional tests to demonstrate compatibility of the hardware with itself and with its environment, and to qualify new or modified designs for flight use.

In general, a development and/or a qualification unit is required for each new design. All qualification and protoflight units are reused for higher level subsystem, panel and STR level tests. The protoflight STR will be refurbished as a flight unit.

Of the 35 types of typical components that are used on an STR, 13 are new, 3 are modifications of existing designs and 19 are existing designs.

4.6.2 COMPONENT TEST

A review of equipment heritage, including current design status of typical flight hardware components and prior use of the components, was used to identify the new or modified components that would require development and/or qualification testing. It was assumed that the components that have been qualified for previous use will be qualified for use on the STR by similarity. Preliminary thermal and acoustic analyses support the assumption. Further analysis and the results of structural development tests will be used to confirm the components' qualification status.

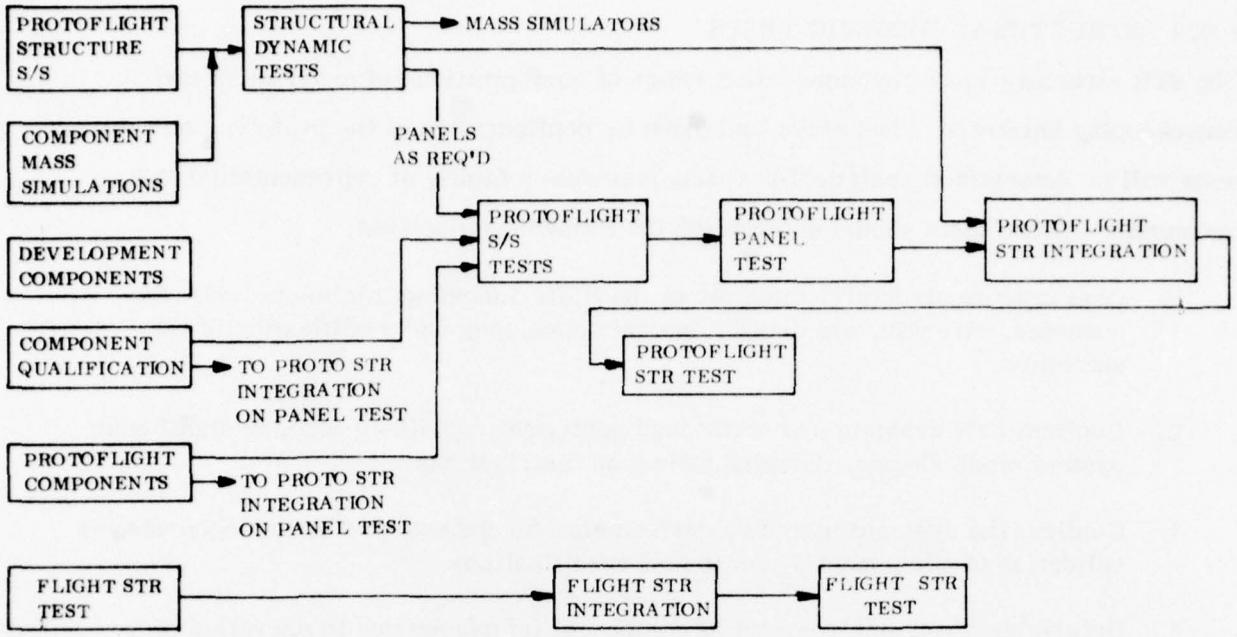


Figure 4.6-1. STR Test Flow

Equipment heritage data for typical components is shown in Appendix D. The data includes the current design status (new, modified, or existing), and the probable source and prior use of the hardware if known.

Component level environmental test requirements are shown in Appendix D. Functional and environmental test requirements are shown for development, qualification and acceptance testing. Random vibration and thermal vacuum/thermal cycling were selected as the environment screens that would be most effective for component test. EMC/ESD testing will be performed at higher levels of assembly. Analysis of Structural Dynamic Model (SDM) test results will be used to confirm or select the component random vibration test levels.

Hardware use and reuse data is shown in Appendix D. The matrix shows one-time use of engineering model components, mass simulations of components for the SDM, and reuse of qualification components for the protoflight STR.

4.6.3 STRUCTURAL DYNAMIC TESTS

The STR structure must accommodate a range of configurations of experiment and housekeeping hardware. The static and dynamic configuration of the prototype structure tests will be determined analytically to accommodate a family of representative experiments. These tests should accomplish the following objectives:

1. Demonstrate structural integrity of the STR: including minimum modal frequencies, strength, alignment, dynamic envelope, and qualification of the structure.

2. Confirm STR dynamic and static load analytical models by defining significant system mode shapes, damping, and load distributions.

3. Confirm the dynamic interface environment for subsystems and components for validation of component dynamic test specifications.

4. Determine shock environment at component interfaces due to operation of STR pyro devices.

5. Confirm alignment repeatability of component mounting planes.

6. Confirm component qualification status for those components to be qualified by similarity.

The structural development test configuration will consist of prime quality primary structure, secondary structure and panels and mass models of housekeeping hardware and representative experiment hardware. The structure and panel will be refurbished, if necessary, and reused as the protoflight structure.

The Prototflight Structure Test Flow Plan shall include the following:

1. Static Load Tests to:

   a. Qualify the strength, deflection and alignment characteristics of the structure.

   b. Determine local influence coefficients for updating the analytical model.

2. Acoustic Noise Test to:

   a. Verify the integrity of the structure for launch dynamic loading.

   b. Evaluate the adequacy of components dynamic test requirements.

3. Pyrotechnic Shock Test to:

   a. Determine the shock environment at component mounting points.

4. Alignment Checks to:

   a. Assess alignment repeatability by pre- and post-environmental test measurements of relationships between mounting planes.

4.6.4 PROTOFLIGHT SUBSYSTEM TESTS

The Protoflight Communication Subsystem and the Protoflight Attitude Control Subsystem will each be tested prior to integration at the panel level of assembly.

Each subsystem will be mounted on protoflight structure panels and will utilize protoflight harness segments, where possible, to minimize the effects of non-flight hardware on the test results.

The objectives of the subsystem tests will be to:

1. Demonstrate subsystem electrical compatibility with minimum interference from interfacing hardware.

2. Partially verify STR harness design and the accuracy of wiring tables.

3. Verify subsystem performance capability.

4. Verify Attitude Control Subsystem software compatibility.

When subsystem tests are complete, the hardware will be reused for the protoflight panel test.

4.6.5 PROTOFLIGHT PANEL TEST

The protoflight panel test demonstrates the compatibility of electrical components mounted on the housekeeping panel and of "bench" mounted components that interface with them. The hardware items necessary for this test are identified in the hardware usage matrix shown in Appendix D.

The specific objectives of this test are to:

1. Verify subsystem compatibility with interfacing hardware in an STR electrically mated configuration.
2. Demonstrate STR electromagnetic compability.
3. Determine RF compatibility of STR components and obtain STR electrical transient signatures.
4. Integrate, and demonstrate compatibility of the Test Ground Station with the STR.

In addition to the compatibility testing, the housekeeping panel will be subjected to thermal vacuum testing to demonstrate electrical compatibility in the simulated space environment. This will be done with a fixture which simulates the thermal control subsystem in one plane and provides for an experiment mount.

4.6.6 STR INTEGRATION AND TEST

The protoflight and flight STR test flows are Figure 4.6-2.

4.6.7 ASSEMBLE STR

STR assembly will consist of mechanical integration of STR panels and components in the "ship" configuration. The harness will be installed with no external connectors mated. Interior subsystem connectors, previously mated for subsystem on panel tests, will remain connected or will be connected during the assembly process.

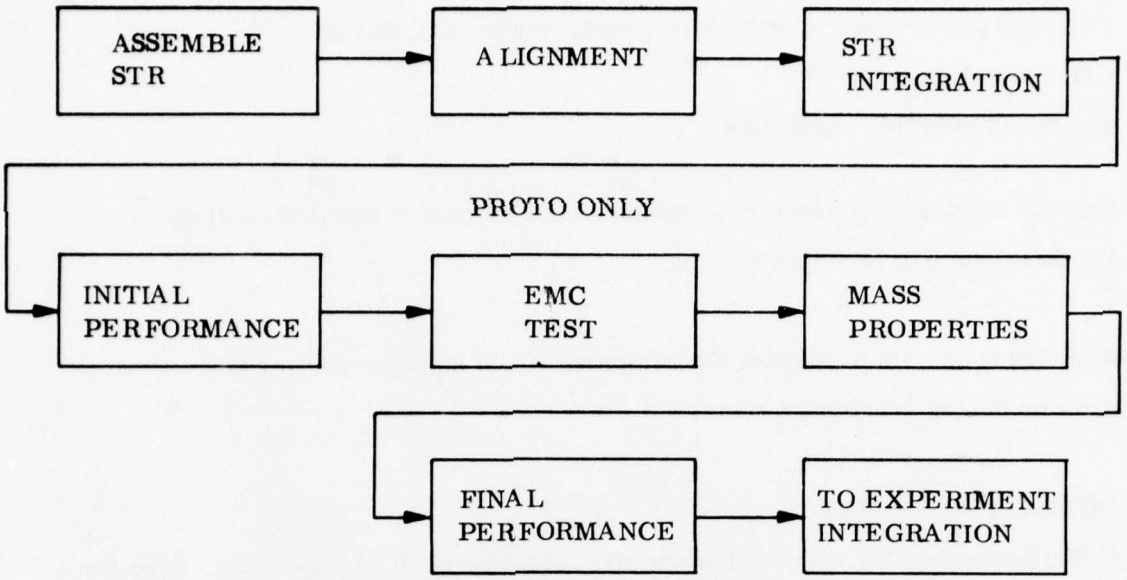


Figure 4.6-2. STR Integration and Test Flow

4.6.8 ALIGNMENT

The alignment repeatability of STR components will have been verified during the structural dynamic test of the protoflight structure. The STR alignment function will be performed once either during the assembly cycle or afterwards just prior to STR integration alignment or the following components will be required:

1. Communication Antennas
2. Sun Sensors
3. Inertial Reference Unit
4. Star Trackers
5. Gimbal Assembly
6. Gyros

4.6.9 STR INTEGRATION

Prior to mating electrical connectors, the following tests will be performed:

1. Unloaded power checks, to assure proper power distribution
2. Ground isolation checks
3. Command override capability

After mating the electrical connectors, command and telemetry functions and caution and warning functions will be checked.

The intent of this testing is to protect the equipment from damage and to assure control of STR functions during subsequent tests.

4.6.10 INITIAL PERFORMANCE

The Initial Performance Test establishes the STR baseline performance data. Specific objectives of this test are to:

1. Demonstrate compatibility between STR subsystem to subsystem interfaces and STR external interfaces with the STR ground station, STR ground support equipment, and simulators, Shuttle and experiment interfaces. The baseline STR interface matrix is shown in Table 4.6-1. The field-of-view and thermal interfaces shown on the table will be verified analytically.
2. Electrical operation of the STR in all of its operating modes and verification of STR functions.
3. Verification of redundant equipment operation.

4.6.11 EMC TEST

The EMC test will perform power transient and bus noise measurements on selected STR signal lines. These values will be compared to predicted values or values measured during component, subsystem, or panel tests. The objective of this test will be to assure that sufficient margin exists for EMI sensitive circuits. The EMC test will be performed on the protoflight STR only.

4.6.12 MASS PROPERTIES

Mass properties of the assembled STR will be measured on the protoflight STR only. These measurements will be used to confirm analytical predictions of mass properties

Table 4.6-1. STR Interface Matrix

2 Pwr S/S	3 Struct S/S	4 Therm S/S	5 ACS	6 Computer	7 STR GSE	8 Shuttle	9 Ground Station	10 Payload	S/S
P, C, Tm	M, Th, Tm, A	Th, C, Tm	C, Tm	C	Es, Tm, C	Es, F	C, Tm, Es	C, Tm, Es	1 Comm S/S
	M, Th P	Th, P	P	P	P, Es	Es	—	P	2 Pwr S/S
		Th, M	Th, M, A	M, Th	M	M, Th	—	M, Th	3 Struct S/S
			Th	Th	—	—	—	Th	4 Therm S/S
				Es	—	F	—	—	5 ACS
					—	—	—	—	6 Computer
						—	—	—	7 STR GSE
							—	—	8 Shuttle
								—	9 Ground Station
									10 Payload
									11

KEY

F - Field of View
A - Alignment
M - Mechanical
Th - Thermal
C - Command
P - Power
Tm - Telemetry
Es - Electrical
S/S - S/S Peculiar
— - Not Applicable

and assure that the analytical technique will be adequate for flight STR mass property determination. Mass properties of flight STR's will be determined from measured weights for each different STR configuration.

4.6.13 FINAL PERFORMANCE

The Final Performance test will be a repeat of the Initial Performance Test. On the protoflight STR, the data will be compared for repeatability with the Initial Performance Test results. The Final Performance test establishes the STR baseline performance for flight STR's.

4.7 STR PROGRAM WORK BREAKDOWN STRUCTURE

The STR Program consists of the following categories of activity:

1. Design, development, test and fabrication of the basic STR system which will support a variety of payloads. This is a one time activity.

2. Integration of a payload into the STR. This consists of STR/Payload assembly design, payload integration and test. This is an activity cycle that is repeated for each payload assembly.

3. Integration of the STR/ Payload Assembly into the STS. This activity is described in Section 5. This is an activity cycle that is repeated for each flight of an STR/Payload assembly.

4.7.1 STR DESIGN DEVELOPMENT TEST AND FABRICATION

The STR design, development, test and fabrication activity includes the following basic items:

1. Systems engineering and analysis of STR and an envelope payload
2. Design of STR
3. Tests including:
   a. Structural development (dynamic, static load, acoustic)
   b. Thermal control (thermal balance)
   c. Qualification of the STR system
      - Dynamic
      - Acoustic
      - Thermal Vacuum
      - Ambient electrical systems
      - RF development
      - EMI development
4. Hardware fabrication for flight usage
5. MAGE design and fabrication
6. EAGE design and fabrication
7. Ground station simulator design & fabrication
8. Operational Site definition
9. STR system usage training

A work breakdown structure for this effort is shown in Figures 4.7-1 and 4.7-2.

4.7.2 STR PAYLOAD INTEGRATION

The STR/ Payload integration activity includes the following basic items:

1. System Engineering and Analysis. This is an update of the general STR analyses applicable to a specific experiment and its interfaces with STR.

2. Design of the STR/ Payload Assembly. Primarily drawing production.

3. Tests. STR/ Payload interface using the STR simulator and the STR mounting panel on which the payload will actually fly. This is usually done at the payload contractor's factory during his development cycle. It insures that at delivery, the payload and the STR will be compatible.

4. Integration and Test of the STR/ Payload Assembly. This is done at the launch site, ETR or WTR.

5. Preparation of mission/payload software.

A work breakdown structure for this activity is given in Figure 4.7-3.

4.7.3 INTEGRATION OF STR/PAYLOAD ASSEMBLY

The Integration of the STR/payload assembly into the STS includes the following basic items:

- Definition and provision of payload support and mission equipment (provided by the payload contractor).

- Definition and provision of STR mission equipment such as aft flight deck consoles, harnesses, caution and occurring interferences hardware.

- Interface with the AFSCF or NASA STDN communications systems.

- Pre-installation (into Orbiter) and post-installation orbiter and payload interface compatibility and verification tests. Support of the installation of the STR/experiment into CITE, PPR, PCR and the Orbiter is included in this task.

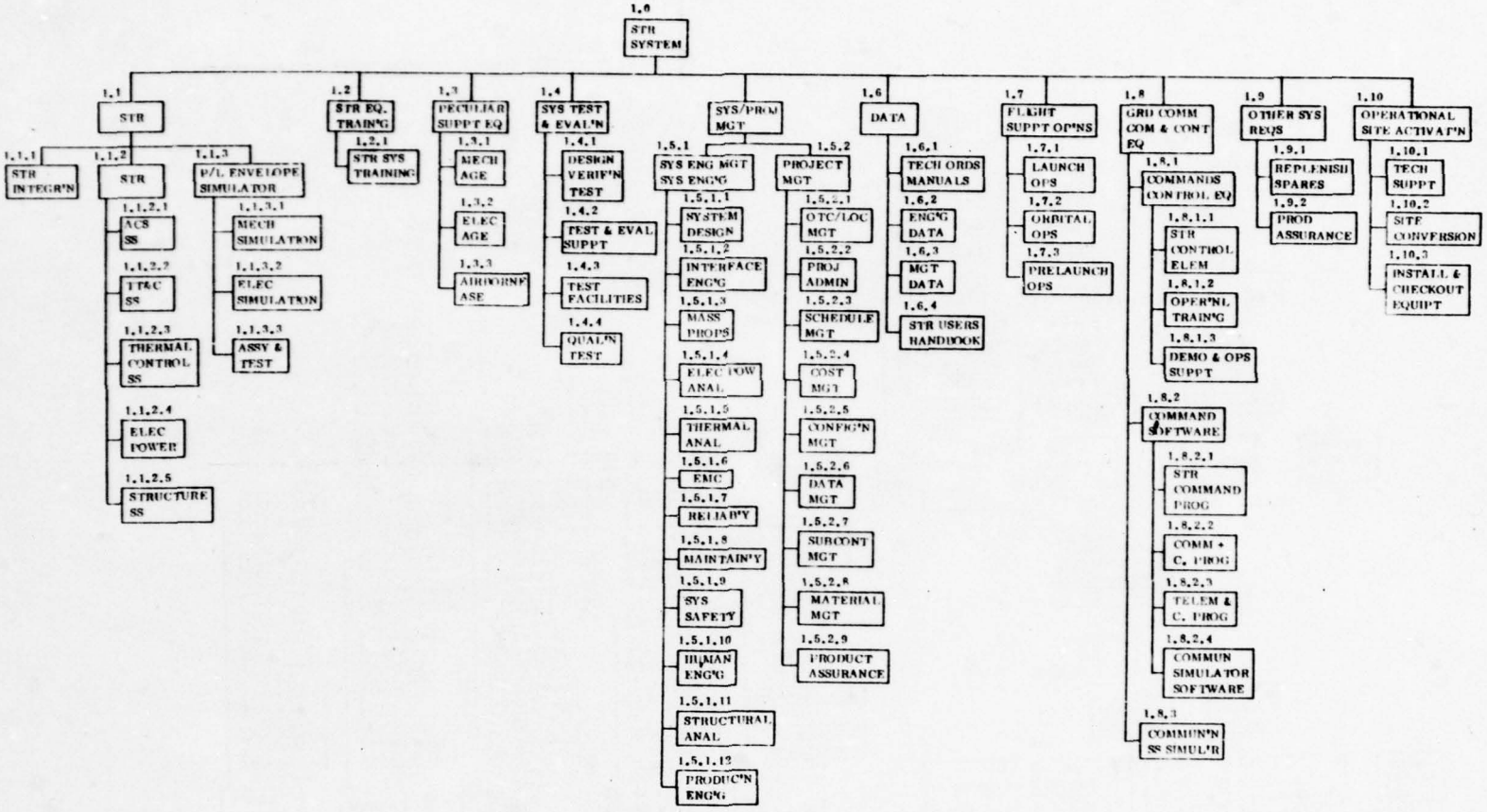


Figure 4.7-1. STR Work Breakdown Structure

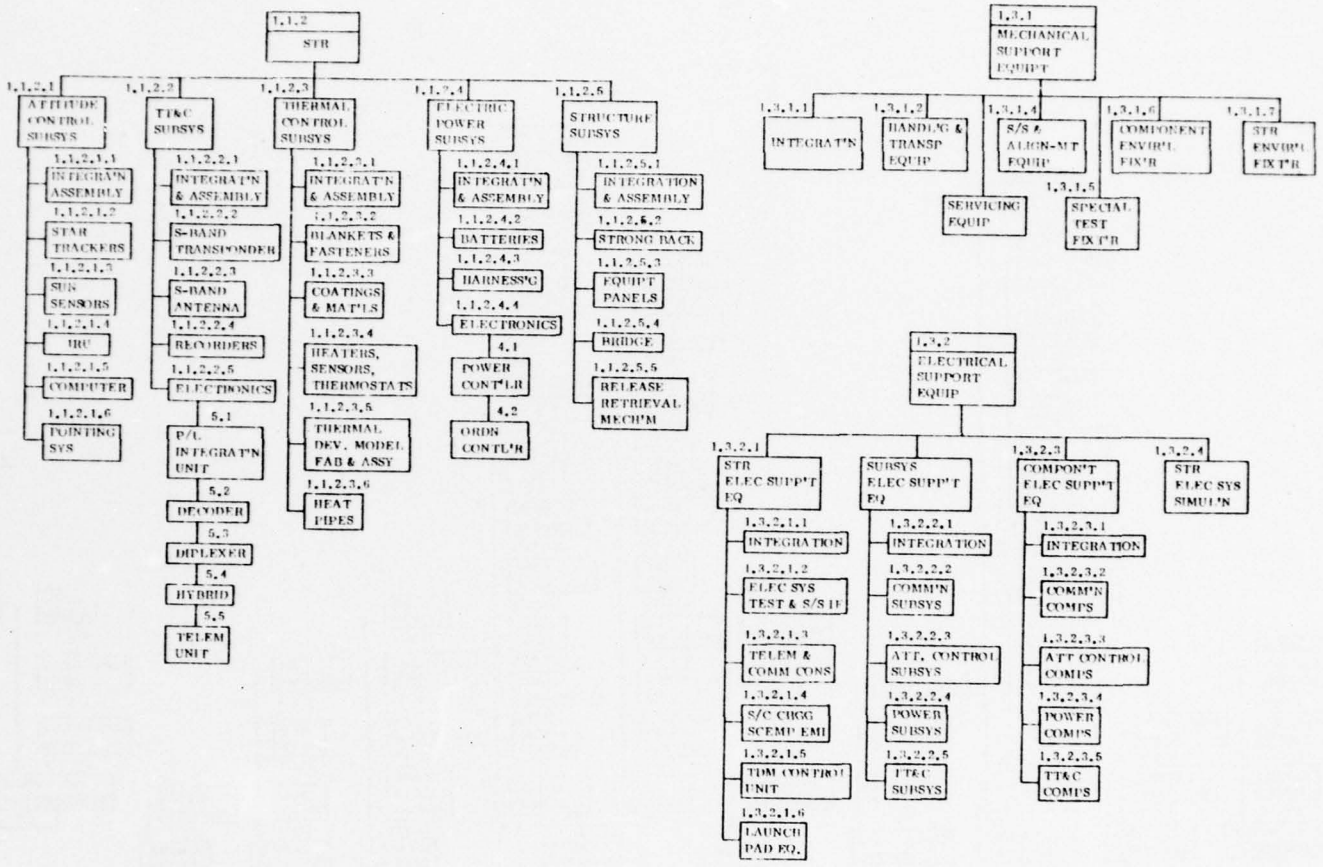


Figure 4.7-2. STR and Support Equipment Work Breakdown Structure Hardware Breakout

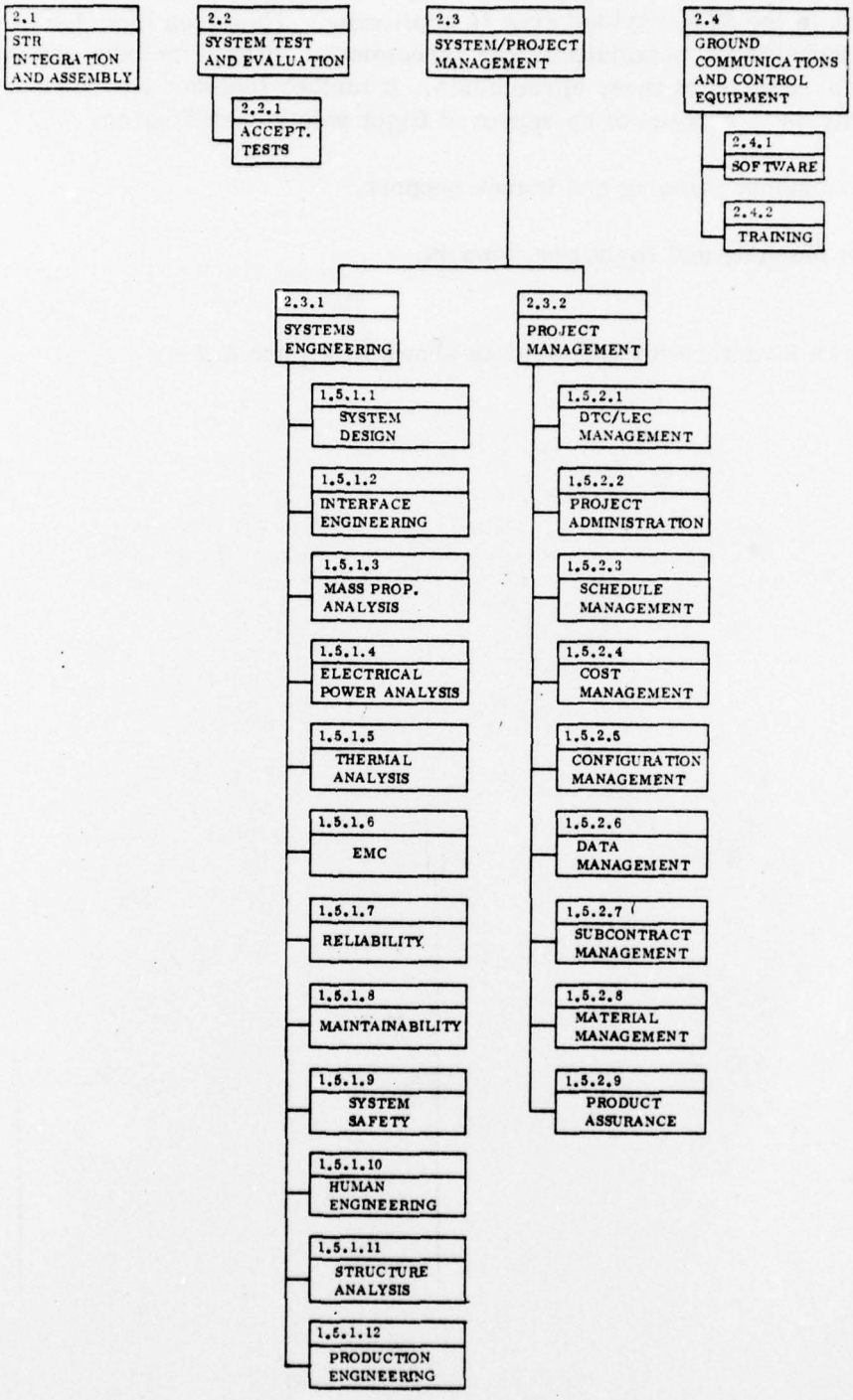


Figure 4.7-3. STR/Experiment Integration Work Breakdown Structure (WBS)

- Systems engineering and analysis for the specific mission flight. Much of the activity in the STR/payload area is applicable. This area includes interface definitions, negotiations and agreements. It also includes the analysis required for support of those agreements. It further includes the hazards and safety analysis in support of an approved flight safety certification.

- Prelaunch support planning and launch support.

- Flight test planning and flight test support.

The Work Breakdown Structure for this area is shown in Figure 4.7-4.

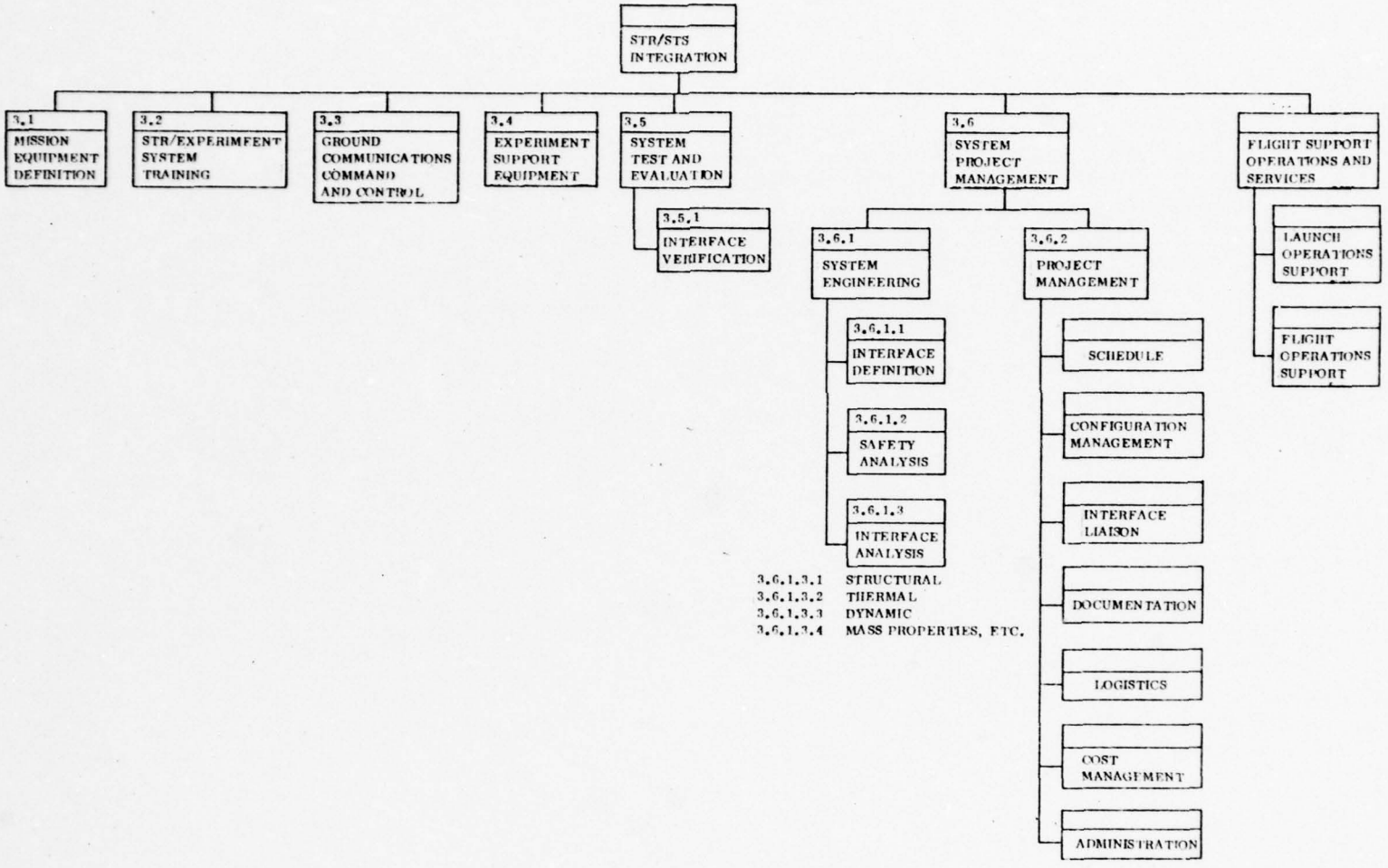


Figure 4.7-4. STS Integration Work Breakdown Structure

SECTION 5

STR OPERATIONAL CONCEPTS (TASK 3)

5.1 INTRODUCTION

5.1.1 BACKGROUND

The DOD Space Test Program (STP) has a continuous program of flight testing experimental equipment as part of proofing that equipment for operational use. Prior to the Shuttle era, such flight tests were carried out on airplanes, balloons, sounding rockets, ballistic missiles, and boosted space flights. Many of the tests were "piggy-back" missions. The Shuttle with the Space Transportation System (STS) adds a new opportunity to fly and recover experiment equipment on a frequent, quick reaction basis. The Standard Test Rack (STR) is a means of implementing this flight test concept on the STS. The design of the STR is discussed in Section 4. The activities required to get the STR with its experiment payload on the Shuttle are the subjects considered in this Section. These activities are analyzed covering the period of time from experiment development thru flight operations, including recovery and re-flight cycles.

5.1.2 SCOPE AND OBJECTIVES

DOD and NASA have conducted a number of studies dealing with the planning of missions for STS and integrating payloads into STS. These study documents, along with the definitive documents defining the STS system, the launch site facilities and other supporting documents defining cost policies, verification test requirements, etc. are the basis for the STR integration analysis. A list of these reference documents is given in paragraph 6.3.1. These documents were reviewed in depth. The data they contain and the STR requirements defined in Section 3 were used to develop STR integration and operations procedures. Previous STS integration studies have dealt with sortie missions and free flight payloads requiring re-flight capability and autonomy respectively. Since the STR is autonomous to the maximum possible extent and never leaves the orbiter it has characteristics similar to both sortie and free flight payloads. The analytical planning activities and the procedures for actual STR

integration into the STS are therefore a combination of both types of STS payloads.

The objectives of the operational concepts task are to:

1. Examine and define the interfaces between the STR and the Shuttle and the launch site facilities. These interfaces include functional, safety, crew participation and the facilities at KSC and VAFB.

2. Define the integration approach to be used for getting STR on the Shuttle.

3. Define the STR activity flows and timelines for accomplishing STS integration with various types of payloads. These include the Spacelab module and pallets in various combinations, spacecraft, and spacecraft with upper stages (IUS, SSUSA or SSUSD).

4. Determine the impact of STR on STS payload integration and define the special activities, equipment and facilities required to accomplish STR integration into STS.

5. Define the key issues and problems to be faced with STR integration.

5.1.3 ASSUMPTIONS AND GUIDELINES

1. STR development is completed prior to first usage.

2. STR can be flown on any STS flight with any cargo configuration consisting of spacelab modules and pallets, spacecraft and spacecraft on upper stages (IUS, SSUS). Length of flight mission is not considered to be a limitation for this part of the study.

3. STR consists of the subsystems which support the experiments. The STR Assembly consists of the STR and the payload which has been integrated onto it.

4. STR is always a secondary payload.

5. The payloads carried by the STR include

   a. Experiments being tested to verify their performance for operational use

   b. R&D operation missions (prime mission high risk component qualification)

   c. Operational user missions

5.2 SHUTTLE INTERFACES

In this section the shuttle interfaces are reviewed and the degree of STR autonomy is assessed.

5.2.1 GENERAL

The STR will interface with the orbiter and other STS payloads which comprise the cargo as shown in Figure 5-1. The interfaces with the other STS payloads are limited to functional compatibility and providing non-interference with installation and flight operation to the maximum possible extent. The STR will also be functionally compatible with the Orbiter. The service accommodations provided by the Oribter are shown in Figure 5-2. These services are provided by the Orbiter at the cargo bay locations shown in Figure 5-3. An analysis of the STR missions and the Shuttle requirements show that STR design provides its own sub-system services and functions (See Section 4) so that only the following interfaces with the Orbiter will exist:

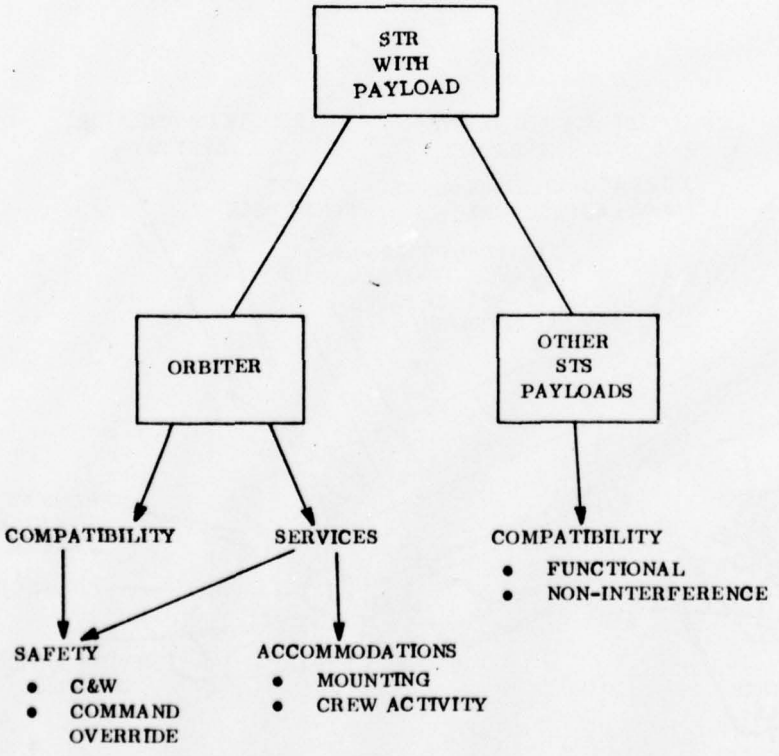


Figure 5-1. STR/STS Interfaces

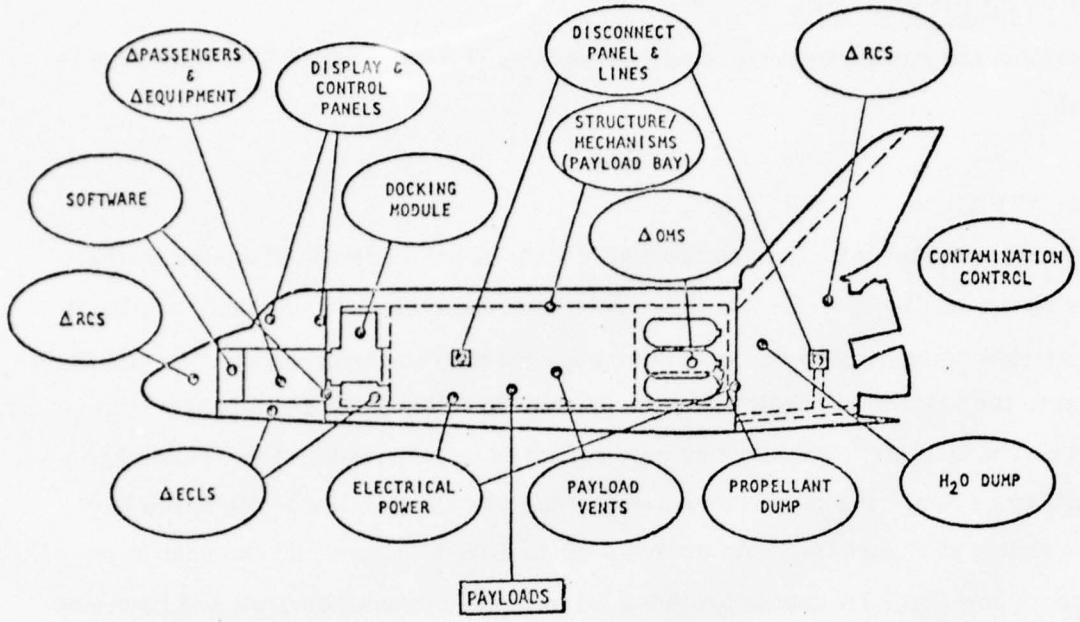


Figure 5-2. Potential Orbiter/Payload Interfaces

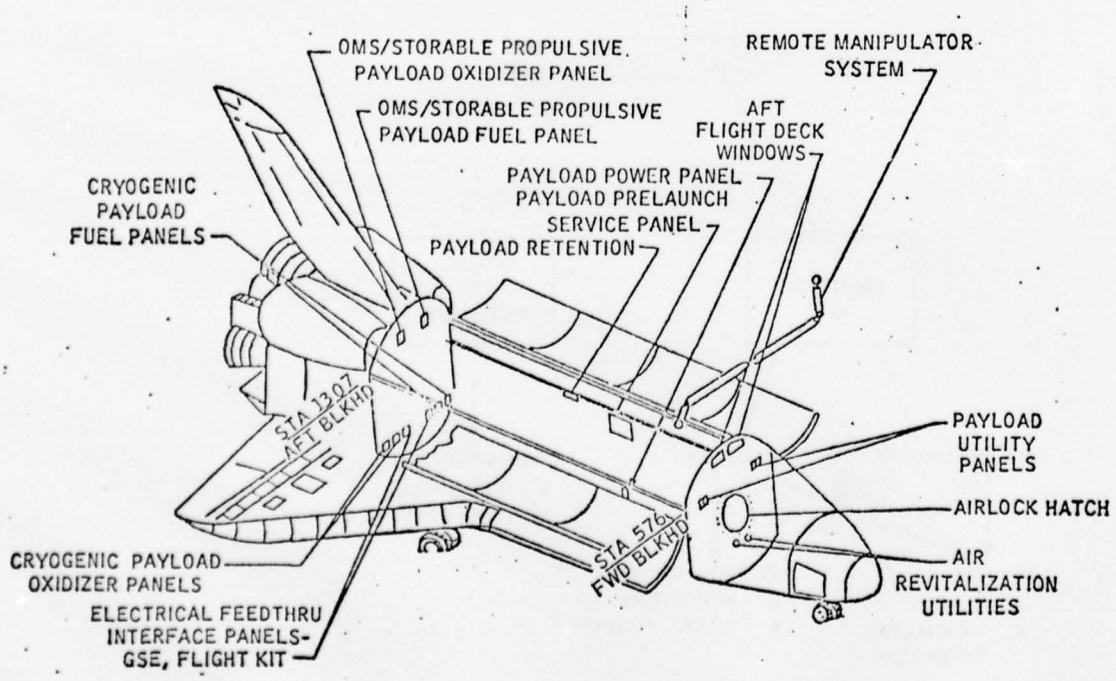


Figure 5-3. Physical Orbiter/Payload Interfaces

1. Physical mounting of the STR on the Orbiter structural support mechanisms.

2. Crew activity consisting of:

   a. STR and experiment operation, limited primarily to turn-on and turn-off.

   b. Operation of the Remote Manipulator System to deploy and retrieve STR payloads.

   c. Other activities such as monitoring performance, target selection, experiment pointing, photography, equipment adjustment etc.

3. Caution and warning requirements for safety purposes are dictated in Reference 1. Current trend on caution and warning is to eliminate the need for it by designing equipment to have high safety margins and for multiple failures to occur before a hazard is created (No single point failure to result in creation of a hazard). However, details of C&W requirements with respect to items such as explosive devices, latches which hold equipment in a stowed position for safety and extendable equipment which impinges into the closed-door (cargo bay door) area have not been fully determined.

4. Safety considerations allowing the crew and/or Shuttle Mission Control Center to override commands to STR. This appears to be a NASA requirement that the Shuttle crew have some control over the cargo. This could mean a turn-on, turn-off capability as a minimum with other potentials such as jettison of STR equipment.

   The twelve experiments used to establish STR baseline requirements were examined and the Orbiter interfaces to be considered were identified. A matrix of the experiments and the four required interfaces is shown in Table 5-1.

5.2.2 MISSION

The STR/Payload missions must be compatible with the primary STS mission. Each STR/Payload mission will have an orbit altitude range, orbit inclination range and attitude duty cycle range. These requirements for each STR/Payload will be compared to the existing STS flight mission definitions specified in the official STS Traffic Manifest. For each STR/Payload, a list of all possible STS flights will be maintained and updated continuously from the time the payload mission requirements are identified. This activity is part of the STR integration planning effort.

Table 5-1. Experiment Interfaces

Interface	Experiment											
	1	2	3	4	5	6	7	8	9	10	11	12
• Physical Mounting	X	X	X	X	X	X	X	X	X	X	X	X
• Crew Activity												
- Turn-on, Turn-off	X	X	X	X	X	X		X	X	X	X	X
- RMS												
Deploy						X				X		X
Retrieve				X	(Possible)	X				X		X (Possible)
- Other	X	X		X	X			X		X	X	
	30%	10%		10%								
• Safety												
Command Override	X	X	X	X	X	X		X	X	X	X	X
• C&W	X	X	X	X	X	X		X	X	X	X	X

As an example the mission for Experiment No. 1, Lasercom Receiver Test, requires that it points at a satellite in synchronous orbit for 30 minutes, twice a day on consecutive orbits. This mission is incompatible with a primary payload requiring continuous earth pointing without using sophisticated and expensive booms to provide the necessary experiment orientation. Typical primary mission payloads which must be assessed as to compatibility with which the STR could fly are shown in Table 5-2. These missions are all low altitude missions with a range from 160mm to 500mm at inclinations varying from 28.5. to polar. It is anticipated that meaningful data for some experiments can be acquired on flight missions as short as 1 day or multiple 3 day missions. However, a review of the 11 baseline experiments and a number of NASA experiments (see Reference 2) indicates that missions of 7 days total or greater are required. Since most of the Sortie and combined missions from Table 5-2 are 7 days or greater (up to 30 days), these missions are the ones on which the STR/Experiments are most likely to fly. In the case of free fliers with 1-2 day mission times, it is most likely that those short missions would be extended to accommodate a "high priority" STR experiment. As

Table 5-2. Primary Payload Mission Types

Sortie	Free Fliers	Combination
• Spacelab • Module only • Module & pallets • Pallets only • Other • BESS • LDEF	• Spacecraft • Spacecraft/IUS • Spacecraft/SSUS • Multiples of the above	• Spacecraft & pallets • Spacecraft/IUS • Spacecraft/SSUS • Multiples of the above

such, the STR then becomes a primary payload. This study does not differentiate between the STR as a primary on secondary payload, and the goals for integration and operational modes of handling each is considered to be the same, i.e. quick turn-around and minimal interface.

5.2.3 FUNCTIONS PROVIDED TO STR EXPERIMENTS

As explained in paragraph 1.1, the goal of the STR is to be autonomous in operation with respect to the Orbiter. An analysis of experiment requirements and Shuttle limitations was performed and is discussed in Section 3. The results of that analysis and the resulting interfaces are defined in this section. A list of the functions required to support the STR experiments is given in Table 5-3. As shown in Table 5-3, the thermal control, power and energy, telemetry and attitude control are provided to the experiments by the STR. No support from the Orbiter is required and a clean interface between the Orbiter and the STR exists in these areas.

Table 5-3. Functions Provided to STR Payloads

	STR	Orbiter	STR/Orbiter Physical Interface
Structure	X		X
Thermal	X		
Power	X		
Telemetry	X		
Command	X		X*
Attitude Control			
Pointing	X	X	
Knowledge	X		
Caution & Warning	X	X	X
Safety	X	X	X
Crew Participation		X	X

*STR/Orbiter command interface is a function of safety and/or crew participation in the STR/Payload mission.

A specific structural interface between the STR and Orbiter always exists since the STR must be mounted in the Orbiter and must survive the static and dynamic structural loads during all mission phases. Details are discussed in paragraph 4.5.

A command interface is tied in with the safety interface because some control over the Orbiter cargo is required by the crew. Details are discussed in paragraph 5.2.6. Caution and warning and safety requirements for the Orbiter as specified in the Space Shuttle System Payload Accommodations document (Reference 1) may also require a physical interface between the STR and the Orbiter. Details are discussed in paragraph 5.2.5 and 5.2.6. Eleven of the twelve baseline experiments required crew participation from a functional standpoint. The functional and safety aspects of crew participation are discussed in paragraphs 5.2.6 and 5.2.7.

5.2.4 MECHANICAL

The mechanical interface with the Orbiter consists primarily of the payload bay location, physical attachment to the Orbiter, c.g. effects on overall cargo limits and non-interference with other payloads.

5.2.4.1 STR Configurations

The basic configurations of the STR are shown in Figure 4.5-1 in Section 4. These configurations provide for support of:

1. A wide range of experiment weights and sizes
2. Payloads on an STR supported pointing system
3. Location of payloads in available space on flights where the primary payload is Spacelab or free fliers (spacecraft which are deployed from the Orbiter).

The basic STR configuration is shown in Figure 5-4. It consists of the strongback (lower 6 panel section) and the bridge (horizontal 3 panel beam). The trunnions which support the STR on the side longerons and the keel are mounted on the strongback. The strongback also contains the STR housekeeping systems (power, telemetry, command, data handling, attitude control, thermal control, etc.). Experiments are mounted on the bridge along with necessary housekeeping elements requiring a visual or RF field-of-view (startracker, sun sensor, S-band antennas). The other configurations are variations of the bridge which supports the experiments. For experiments which require large vertical direction mounting, lower bridge

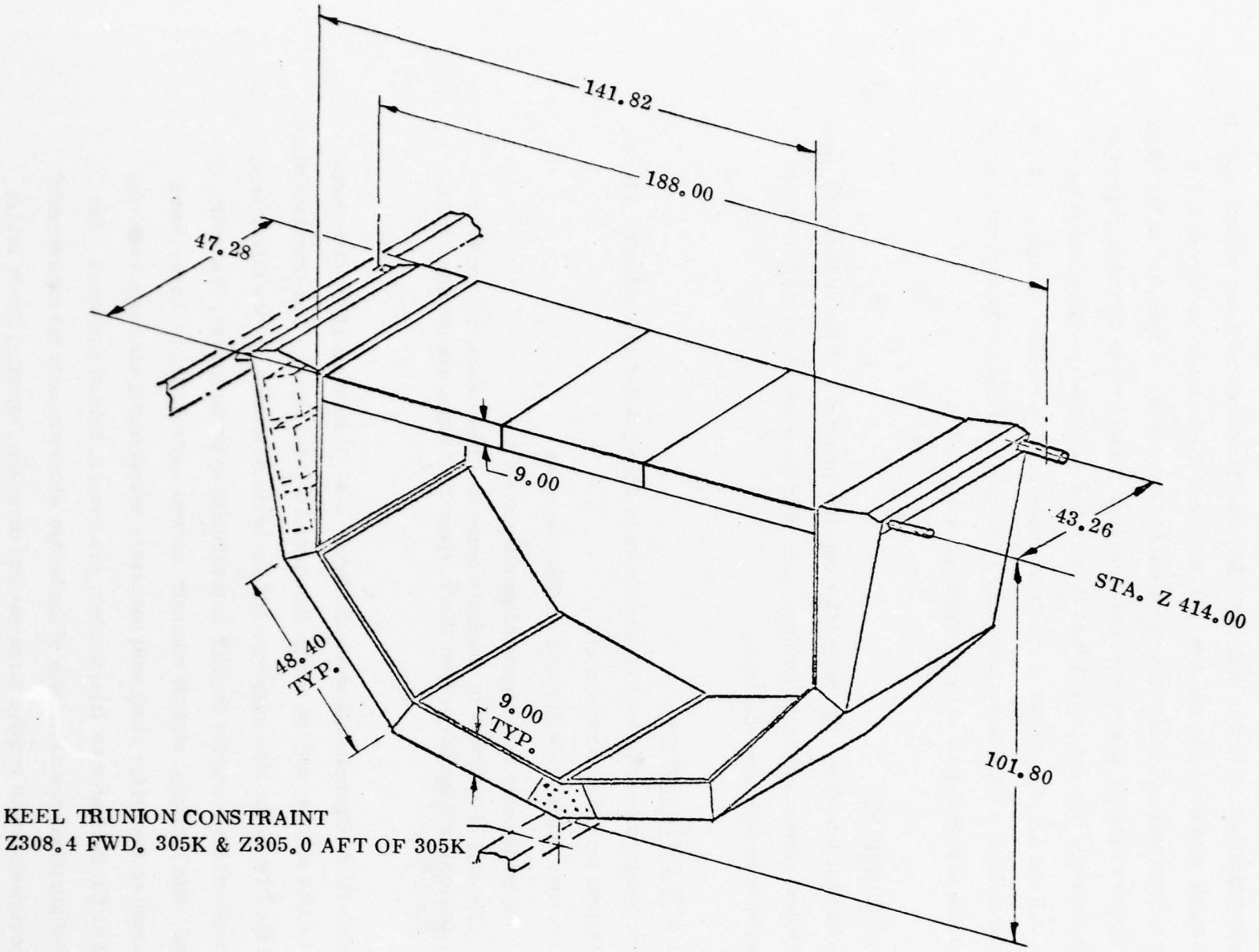


Figure 5-4. STR Basic Configuration

positions are available. As an example, a lower bridge configuration can be used for an experiment mounted on a pointing system. (See Figure 4.5-1, Section 4). The ultimate in available volume is provided by using the strongback alone. In this case payloads would be mounted directly to the structural hard-points which support the bridge.

For cases where a complete bridge is not possible, the double or single shelf configuration is used. This is utilized, for example, when the STR is mounted around a spacecraft.

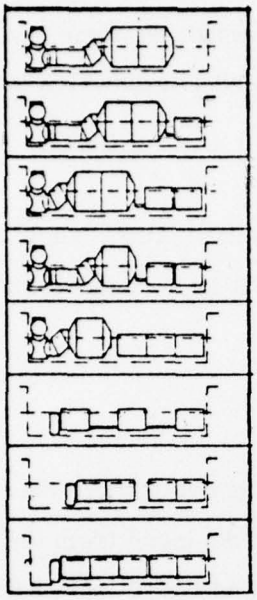
5.2.4.2 STR Mounting Locations

The various prime payloads with which the STR will fly on the orbiter are the spacelab, non-spacelab sortie missions and the free fliers (spacecraft deployed from the orbiter, with and without the upper stages, IUS and SSUS). The first 44 operational STS flights as defined by NASA in Reference 3 were reviewed. The numbers and types of flights include, 19 spacelab, 21 spacecraft, mostly with upper stages and 4 combination spacecraft and non-spacecraft sortie missions. The 7 standard spacelab configurations are shown in Figure 5-5. Typical spacecraft and mixed sortie-spacecraft orbiter payloads are shown in Figures 5-6 thru 5-9.

Orbiter payload types include:

1. Spacelab
2. Spacecraft
3. Spacecraft on IUS
4. Spacecraft on SSUS
   a. Single
   b. Stacked
5. Non-Spacelab sortie and spacecraft
   a. Without upper stage
   b. With upper stage

Available mounting areas include open areas between other primary payloads and between payloads and the fore and aft cargo bay bulkheads, the area around the spacelab tunnel connecting the Spacelab and the orbiter cabin and areas around spacecraft mounted on their own support cradles or on upper stages such as IUS. From the



1 LONG MODULE

2 LONG MODULE, ONE PALLET

3 LONG MODULE, TWO PALLETS

4 SHORT MODULE, TWO PALLETS

5 SHORT MODULE, THREE PALLETS

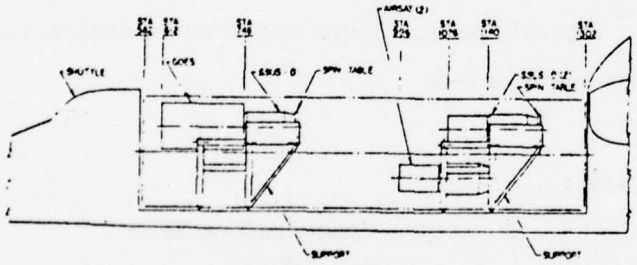
6 THREE PALLETS

7 FOUR PALLETS

8 FIVE PALLETS

Figure 5-5. Standard Spacelab Configurations

MISSION NO. 9

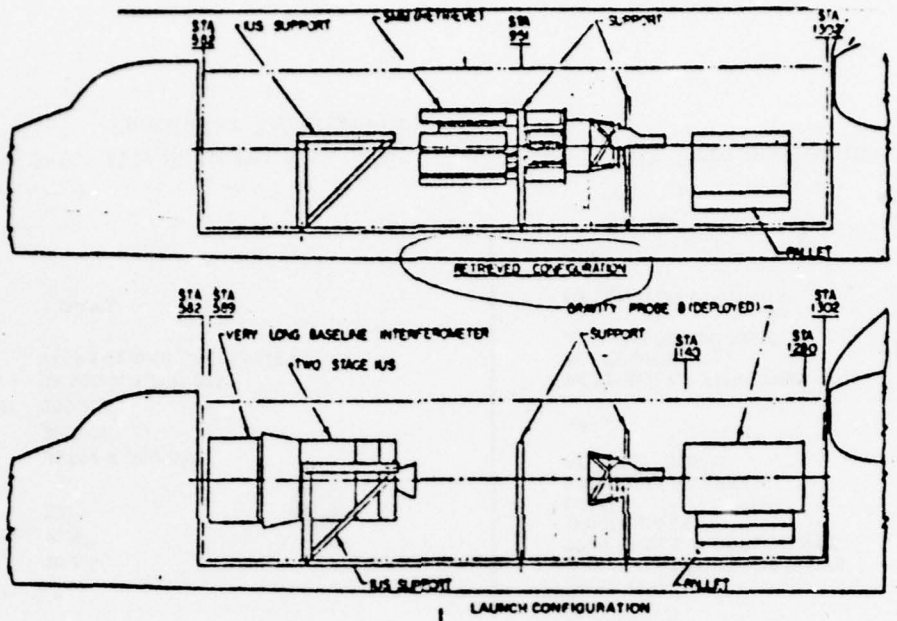


MISSION	SSUS-D WITH AEROSAT SSUS-D WITH AEROSAT SSUS-D WITH GOES
LAUNCH DATE	SEPTEMBER 1980
INCLINATION	0°
ORBITAL ALTITUDE	35,786 X 296 KM
CREW REQUIREMENTS	(0) PAYLOAD SPECIALISTS
MISSION DURATION	~1 DAY

WEIGHT SUMMARY Kg

ITEM	LAUNCH	DOWN
SSUS-D (2)	3510	0
CRADLE STRUCTURE & SYS	1701	1701
SPIN TABLE & DEPLOY SYS	522	522
UMBILICALS	14	14
MISC.	27	27
PAYLOAD ADAPTER (2)	128	0
AEROSAT (2)	1800	0
SSUS-D	1755	0
CRADLE STRUCTURE & SYS	1701	1701
SPIN TABLE & DEPLOY SYS	261	261
UMBILICALS	7	7
MISC	27	27
PAYLOAD ADAPTER	53	0
GOES	630	0
RETENTION FITTINGS	210	210
PSS	25	25
TOTAL	12,371	4495

Figure 5-6. SSUS-D With AEROSAT (2)
SSUS-D With GOES



LAUNCH CONFIGURATION

MISSION
IUS (2 STAGE) WITH VERY LONG BASELINE INTERFEREMETER
GRAVITY PROBE B– DEPLOY
SMM–RETRIEVE
APPS

LAUNCH DATE
OCTOBER 1981

INCLINATION

VERY LONG B/L INT	:	—
GRAVITY PROBE B	:	33.0°
SMM	:	33.0°

ORBITAL ALTITUDE

VERY LONG B/L INT	:	C_3 = 26 KM^2/SEC^2
GRAVITY PROBE B	:	460 KM
SMM	:	460 KM

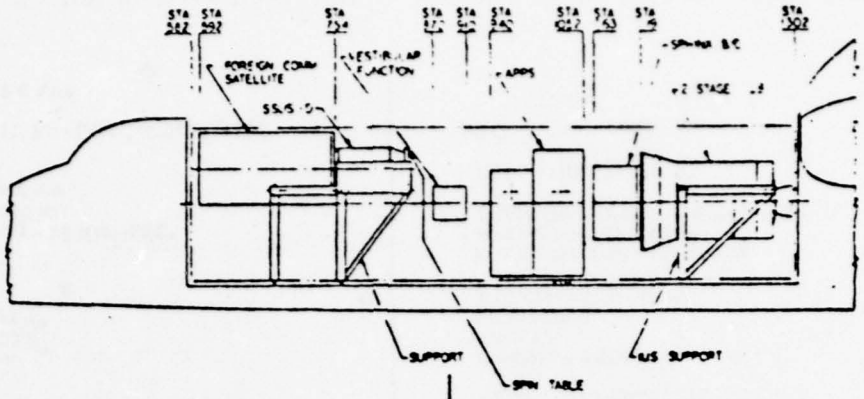
CREW REQUIREMENTS	:	(1) PAYLOAD SPECIALISTS
MISSION DURATION	:	7 DAYS

WEIGHT SUMMARY – Kg

ITEM	LAUNCH	DOWN
FIRST STAGE IUS @ IGN	10395	0
SECOND STAGE IUS @ IGN	3449	0
SHUTTLE INTERFACE (IUS)	1657	1657
PAYLOAD ADAPTER	64	0
VERY LONG BASELINE INTER	907	0
GRAVITY PROBE B	650	0
PALLET	600	600
PLATFORM	95	95
ATTACH STRUCTURE	80	80
SOLAR MAXIMUM MISSION	0	1590
ATTACH STRUCTURE	864	864
THERMAL CONTROL & PWR DIST	95	95
RETENTION FITTINGS	420	420
PSS	45	45
TOTAL	19321	5448

Figure 5-7. IUS (2-Stage With VLB-1), Gravity Probe B-Deploy SMM-Retrieve

FLIGHT NO. 16



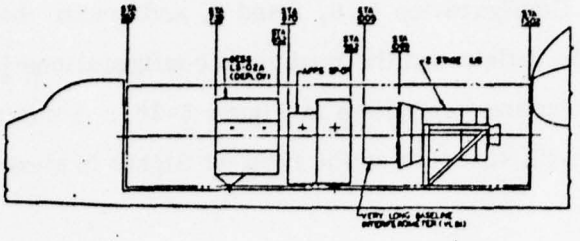
MISSION	SSUS-D WITH FOREIGN COMM. APPS VESTIBULAR FUNCTION-DEPLOY IUS (2 STAGE) WITH SPHINX B/C
LAUNCH DATE	MAY 1981
INCLINATION	
FOREIGN COMM	0°
APPS	28.5°
VESTIBULAR FUNCTION	28.5°
SPHINX B/C	20°
ORBITAL ALTITUDE	
FOREIGN COMM	35786 X 300 KM
APPS	300 KM
VESTIBULAR FUNCTION	300 KM
SPHINX B/C	36,000 X 1,000 KM
CREW REQUIREMENTS	(1) PAYLOAD SPECIALIST
MISSION DURATION	7 DAYS

WEIGHT SUMMARY Kg

ITEM	LAUNCH	DOWN
FIRST STAGE IUS @ IGN.	10395	0
SECOND STAGE IUS @ IGN.	3449	0
SHUTTLE INTERFACE (IUS)	1657	1657
PAYLOAD ADAPTER	38	0
SPHINX B/C	318	0
SSUS-D	1755	0
CRADLE STRUCTURE & SYS	1701	1701
SPIN TABLE & DEPLOY SYS	261	261
UMBILICALS	7	7
MISC	27	27
PAYLOAD ADAPTER	52	0
FOREIGN COMM	600	0
APPS	4784	4784
VESTIBULAR FUNCTION	200	0
PLATFORM	200	200
ATTACH STRUCTURE	27	27
RETENTION FITTINGS	560	560
PSS	50	50
TOTAL	26081	9274

Figure 5-8. IUS (2 Stage) with Sphinx B/C SSUS-D with Foreign Comm APPS, Vestibular Function-Deploy

FLIGHT NO. 41



MISSION	IUS (2 STAGE) WITH VERY LONG BASELINE INTERFEROMETER BESS – DEPLOY APPS
LAUNCH DATE	AUGUST 1982
INCLINATION	
VERY LONG B/L INTER	0°
BESS	28.5°
APPS	28.5°
ORBITAL ALTITUDE	
VERY LONG B/L INTER	5000 KM
BESS	500 KM
APPS	500 KM
CREW REQUIREMENTS	(1) PAYLOAD SPECIALISTS
MISSION DURATION	7 DAYS

WEIGHT SUMMARY - Kg

ITEM	LAUNCH	DOWN
FIRST STAGE @ IGN	10395	0
SECOND STAGE @ IGN	3449	0
PAYLOAD ADAPTER	43	0
VERY LONG BASELINE INTER.	907	0
SHUTTLE INTERFACE (IUS)	1657	1657
APPS	4784	4784
BESS	1818	0
ATTACH STRUCTURE	182	182
RETENTION FITTINGS	250	250
TOTAL	23485	6873

Figure 5-9. IUS (2 Stage) With VLBI Bess-Deploy APPS

survey of the 44 NASA STS payloads, the availability of the various mounting areas with the non-spacelab primary payloads is shown in Table 5-4.

Table 5-4. STR Mounting Opportunities for Non-Spacelab Payloads

Type of Orbiter Payload	Open Areas			Around Spacecraft	Combination of open Areas and Around Spacecraft	No Room	Not Defined
	Forward of Payloads	Between Payloads	Aft of Payloads				
Spacecraft (21 flights)	3			4	4	1	9 (DOD payloads)
Combination sortie and spacecraft (4 flights)	1			2		1	

From Figure 5-5, Spacelabs allow STR mounting around the tunnel in Configurations 1, 2 and 4, in open areas in Configuration 1, 6, 7 and 8, and around the utility, support structure which joins the pallets and the module in configurations 6 and 7. (Details of the utility support structure are shown in Figure 5-10.) A summary of the STR mounting opportunities with spacelab on the first 44 flights is shown in Table 5-5.

Table 5-5. STR Mounting Opportunities With Spacelab Payloads

Total Spacelab Payloads	Open Areas	Around Tunnel		Around Pallet Utility Bridge
		Long Tunnel	Short Tunnel	
19	8*	11	3	1

*Three open areas are in combination with tunnel or the utility bridge.

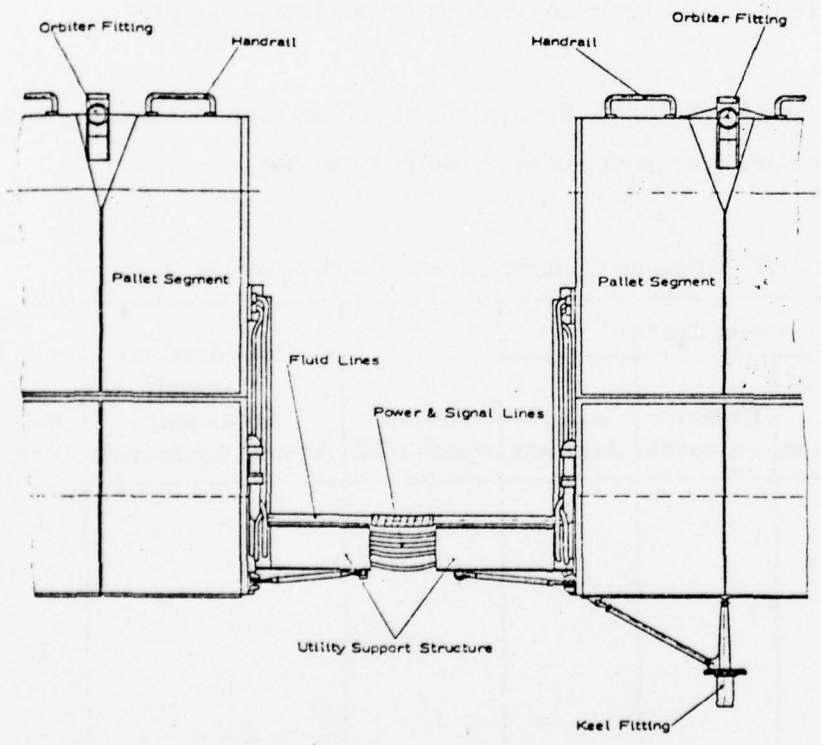


Figure 5-10. Pallet Support Bridge

The number of opportunities for each of the mounting areas mandated the STR design with its alternate configurations to maximize the potential usages of the STR concept. A summary is shown in Table 5-6 below:

Table 5-6. Summary of Types of STR Mounting

Total Number of STS Flights Considered (Multiple Options on Some Flights)	35
STR Mounted	
• Around Spacelab Tunnel	14
• Around Pallet Utility Bridge	1
• Around Spacecraft	6
• In Open Areas	16
• Not Possible	2

The spacelab tunnel is the STR configuration forcing function. The short version of the tunnel with the most recent airlock concept is shown in Figure 5-11. Longer versions of the tunnel permit lower bridge mounting positions. The STR concept can be used with both the short and long versions of the tunnel. The minimum STR length and height available for the tunnel configurations is shown in Table 5-7.

Table 5-7. Tunnel Dimensions

	Length Between Airlock and Spacelab Module, Inches	Height Above Tunnel, Inches	Height Below Tunnel, Inches
Shortest Tunnel	71.18	59.0	16.9
Longest Tunnel	374.04	103.6	16.9

These dimensions include the handrails, flexible couplings and relative deflections of the tunnel from Reference 4.

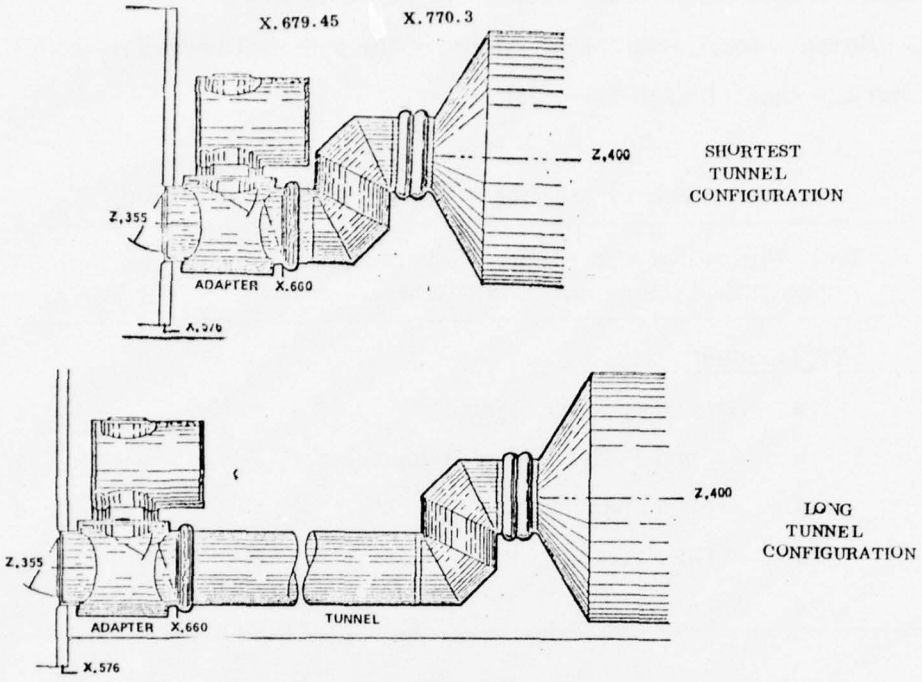


Figure 5-11. Spacelab Tunnel and Airlock Configurations

5.2.4.3 STR Mounting

The STR utilizes the 5-point mounting system as shown in Figure 5-12. This is an acceptable configuration defined in the NASA Space Shuttle System Payload Accommodations document (Reference 1). The STR mounting interface is shown in Figure 5-13.

This five-point system is designed so that the stressed skin, open-U section of the payload bay, including the designs of the Shuttle longerons and ribs, can most efficiently handle the loads generated by the payloads being carried. This leads to the concept that the attachment points on the longerons (rails along each side of the payload bay) should be loaded only in the "plane" of the skin, i.e., fore and aft (±X) and up and down (±Z); whereas, the keel and adjacent rib structures are loaded only in the plane of the skin and along the ribs (±Y) as in Figure 5-13 (upper right). Further, it is desired to isolate the vibrational loads developed in the Orbiter during launch from the payload structure.

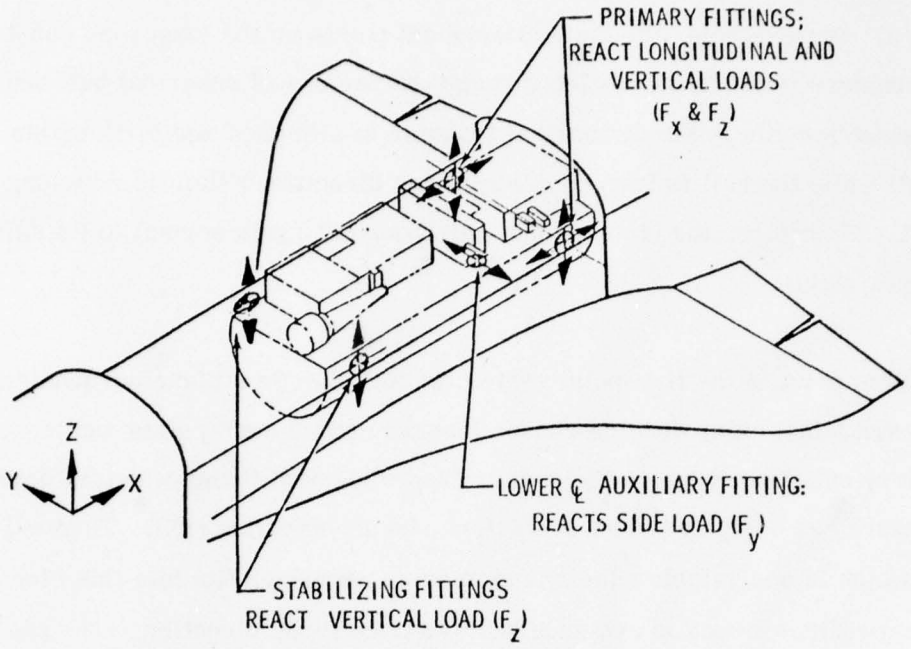


Figure 5-12. 5 Point Payload Retention System (Indeterminate)

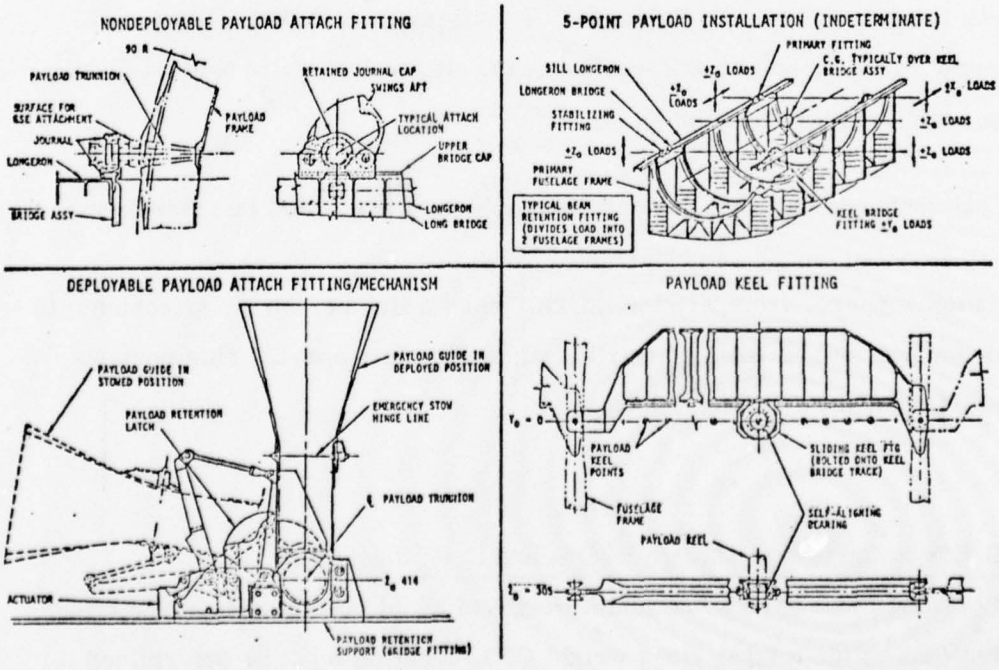


Figure 5-13. STR Mounting Interface

These objectives are met by attachment fittings that hold firm in certain directions and slip in others. For example, the main attachment points on the longerons consist of cylindrical trunnions which fit into holes through the centers of spherical ball bearings. This construction allows the cylindrical trunnion to slip back and forth within the spherical ball while the ball is free to rotate in any direction within its matching spherical socket. Therefore, the trunnion can only transmit loads normal to its axis (Figure 5-13 upper right).

The keel attachment point in the five-point system is coplanar (in a plane perpendicular to the Orbiter longitudinal axis) with the chosen longeron attachment points and is designated to carry only transverse (±Y) loads. The attachment fitting which is bolted to the keel structure has elongated slots in the fore and aft direction (±X). The keel reaction fitting which is equivalent to the trunnion has a peg which fits into this slot, loosely in the longitudinal direction and snugly in the transverse direction. The peg is sufficiently long so that it remains in the hole. Any pitch moments due to the STR CG lying above or below the plane of the trunnions or by other causes is reacted to the payload structure through a pitch arm. The pitch arm supports a trunnion similar to the longeron trunnions, but the pitch arm is free to move fore and aft in a slot, while held snugly in the up and down direction (±Z). In the five-point system there are two pitch arms (or pitch trunnions).

The Orbiter bay longeron and keel payload support points are listed in Table 5-8.

The STR trunnion supports are spaced at 43.26 inches which permit its attachment to the Orbiter at any pair of available supports spaced 11 points apart. This permits maximum flexibility of STR mounting location.

5.2.4.4 STR Effects on Orbiter Cargo Weight and C.G. Location

The STR and payloads will have a maximum weight of about 5000 pounds. The effect of this added weight on the total payload weight C.G. location must be determined for

Table 5-8. Payload Attach Points - Longeron and Keel Xo Locations - Inches (mm)

Attach Pt #	Inches	mm	Attach Pt #	Inches	mm	Attach Pt #	Inches	mm	Attach Pt #	Inches	mm
154K	608.80	(15463.5)	197	777.93	(19759.5)	240	947.07	(24055.5)	282	1112.27	(28251.6)
155K	612.73	(15563.4)	198	781.87	(19859.4)	241	951.00	(24155.4)	283	1116.20	(28351.5)
156*	616.67	(15663.3)	199	785.80	(19959.3)	242	954.93	(24255.3)	284	1120.13	(28451.4)
157	620.60	(15763.2)	200	789.73	(20059.2)	243	958.87	(24355.2)	285	1124.07	(28551.3)
158	624.53	(15863.1)	201	793.67	(20159.1)	244K	962.80	(24455.1)	286	1128.00	(28651.2)
159L	628.47	(15963.1)	202	797.60	(20259.0)	245K	966.73	(24555.0)	287	1131.93	(28751.1)
160L	632.40	(16063.0)	203L	801.53	(20358.9)	246K	970.67	(24654.9)	288L	1135.87	(28851.0)
163L	644.20	(16362.7)	204L	805.47	(20458.9)	247L	974.60	(24754.8)	289L*	1139.80	(28950.9)
164L	648.13	(16462.6)	207L	817.27	(20758.6)	250L	986.40	(25054.6)	292L	1151.60	(29250.6)
165	652.07	(16562.5)	208	821.20	(20858.5)	251L	990.33	(25154.5)	293	1155.53	(29350.5)
166	656.00	(16662.4)	209	825.13	(20958.4)	252	994.27	(25254.4)	294	1159.47	(29450.5)
167	659.93	(16762.3)	210	829.07	(21058.3)	253	998.20	(25354.3)	295	1163.40	(29550.4)
168K	663.87	(16862.2)	211	833.00	(21158.2)	254	1002.13	(25454.2)	296	1167.33	(29650.3)
169K	667.80	(16962.1)	212	836.93	(21258.1)	255	1006.07	(25554.1)	297	1171.27	(29750.2)
170K	671.73	(17062.0)	213	840.87	(21358.0)	256	1010.00	(25654.0)	298	1175.20	(29850.1)
171K	675.67	(17161.9)	214	844.80	(21457.9)	257	1013.93	(25753.9)	299*	1179.13	(29950.0)
172K	679.60	(17261.8)	215	848.73	(21557.8)	258	1017.87	(25853.8)			
173K	683.53	(17361.7)	216	852.67	(21657.7)	259	1021.80	(25953.7)	304*	1198.80	(30449.5)
177L	699.27	(17761.4)	217L	856.60	(21757.6)	260	1025.73	(26053.6)	305	1202.73	(30549.4)
178L	703.20	(17861.3)	218L	860.53	(21857.5)	261	1029.67	(26153.5)	306	1206.67	(30649.3)
179	707.13	(17961.2)	221L	872.33	(22157.3)	262L	1033.60	(26253.4)	307	1210.60	(30749.2)
180	711.07	(18061.1)	222	876.27	(22257.2)	263L	1037.53	(26353.3)	308	1214.53	(30849.1)
181	715.00	(18161.0)	223	880.20	(22357.1)	266L	1049.33	(26653.1)	309	1218.47	(30949.1)
182	718.93	(18260.9)	224	884.13	(22457.0)	267	1053.27	(26753.0)	310	1222.40	(31049.0)
183	722.87	(18360.8)	225	888.07	(22556.9)	268	1057.20	(26852.9)	311	1226.33	(31148.9)
184	726.80	(18460.7)	226	892.00	(22656.8)	269	1061.13	(26952.8)	312	1230.27	(31248.8)
185	730.73	(18560.6)	227	895.93	(22756.7)	270	1065.07	(27052.7)	313	1234.20	(31348.7)
186K	734.67	(18660.5)	228*	899.87	(22856.6)	271	1069.00	(27152.6)	314L	1238.13	(31448.6)
187K	738.60	(18760.4)	229K	903.80	(22956.5)	272	1072.93	(27252.5)	315L	1242.07	(31548.5)
188L	742.53	(18860.3)	230K	907.73	(23056.4)	273	1076.87	(27352.4)	316L	1246.00	(31648.4)
189L	746.47	(18960.3)	234L	923.47	(23456.0)	274	1080.80	(27452.3)	317K	1249.93	(31748.3)
192L	758.27	(19260.0)	235L	927.40	(23556.0)	277L	1092.60	(27752.0)	322L	1269.60	(32247.8)
193L	762.20	(19359.9)	236L	931.33	(23655.9)	278L	1096.53	(27851.9)	323L	1273.53	(32347.7)
194	766.13	(19459.8)	237	935.27	(23755.8)	279L	1100.47	(27951.9)	324L	1277.47	(32447.6)
195	770.07	(19559.7)	238	939.20	(23855.7)	280	1104.40	(28051.8)	325L	1281.40	(32547.6)
196	774.00	(19659.6)	239	943.13	(23955.6)	281	1108.33	(28151.7)	330L	1303.00	(33096.2)

Yo and Zo Locations Inches (mm)

Longeron: Yo ± 94.0 inches (±2387.6 mm), Zo 414.0 inches (10515.6 mm) except for att. pt. 330L, Zo 409.0 inches (10388.6 mm), Yo ± 91.4 inches (±2321.6 mm)

Keel: Forward of Attach Point 304K: Yo = 0, Zo 305.0 inches (7747.0 mm)
Attach Point 304K and Aft: Yo = 0, Zo 308.4 inches (7933.4 mm)

Notes: All attach points normally available are included in this list.
Attach points designated 'L' (e.g. number 160L) are available only at the longerons.
Attach points designated 'K' (e.g. number 168K) are available only at the keel.
*The longeron attach points at these locations may be used only for primary (bolt-down) attachment.

each potential STS flight. If the added STR weight does not permit the orbiter to reach the planned orbit altitude and inclination based on the performance curves in Reference 1 then that flight is not a potential mission flight for STR. A similar effect in C.G. location must also be made for the cargo C.G. limits of Reference 1. An example of the effect of STR on C.G. is as follows: The longitudinal (X axis) locations for the 8 spacelab configurations are shown in Figure 5-14 superimposed on the orbiter payload limit curves. Configuration 5 is the most forward C.G. case. This is the configuration specified in the NASA flight manifest (Reference 3) for STS Flight 21. The spacelab weighs approximately 16,000 pounds (payload is conservatively neglected) with its C.G. at Station 890. Adding a 5000 pound STR payload at the most forward location around the tunnel would put its C.G. at station 704. The net result is a forward movement of the total cargo payload to station 844. This C.G. location is about 15 inches aft of the C.G. limit joint of 830 inches. The total mass of the STR for weight and C.G. purposes will include the STR experiments, wiring for command interface and C&W, and dedicated consoles in the aft flight deck. A survey of the first 44 flights

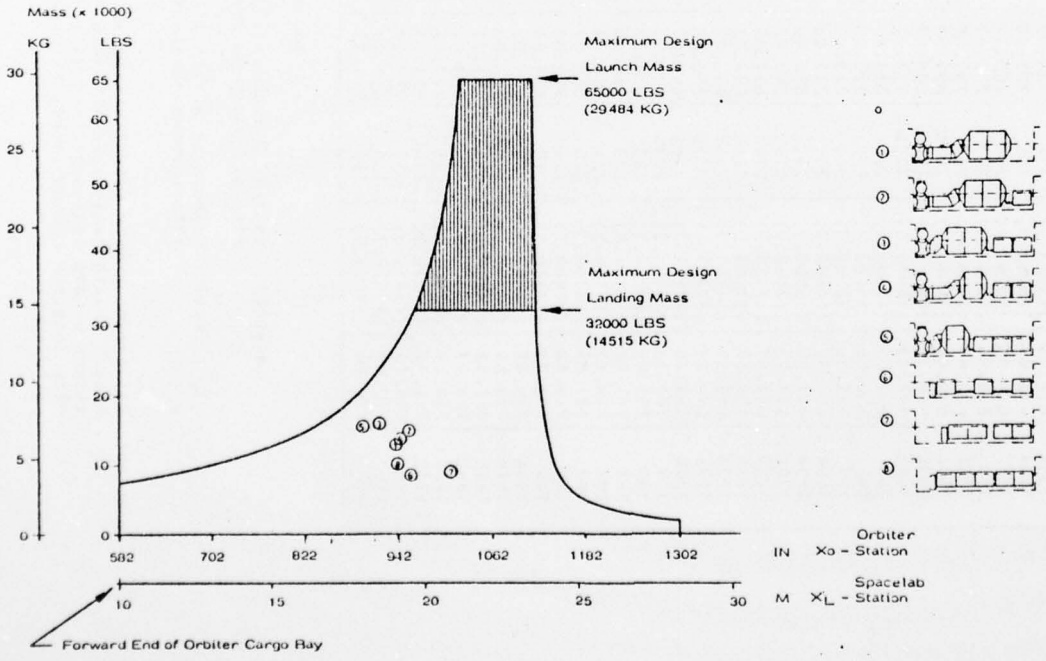


Figure 5-14. X-Axis CG Location of Basic Spacelab Configurations

indicates that the STR can be added to most of them without becomming a limitation to the planned flight mission.

5.2.4.5 Separation and Extension of STR and Payloads

No separation or extension of the total STR was required by any of the 12 DOD baseline payloads. As shown in Table 4-9, three of the baseline experiments are separated from the orbiter and two are extended from the orbiter.

Table 5-9. Separation and Extension of STR Experiments

	Separation Required	Extension Required
Experiment Number	4 (10 Items) 6 12	10 11

One of the three separable experiments, Number 4, has its own separation mechanism (it is part of the experiment). The other two are separable spacecraft, one of which can be retrieved.

Experiments 10 and 11 have similar requirements, such as distance from orbiter and controllable pointing, which make it convenient to use the Orbiter Remote Manipulator System (RMS). The RMS and its reach capabilities are shown in Figures 5-15 and 5-16. Possible locations of STR with separable payloads is defined by Figures 5-15 and 5-16. Only positions in the extreme aft portion of the Orbiter beyond station 1187 and a small area around the RMS mounting point cannot be reached by the Orbiter RMS.

Since the separable STR Payload Number 6 has a retrieval requirement, the RMS would be used for separation since it is the only means for retrieval. No experiment unique separation system would be required. The use of RMS does not affect the time-lines for installation of STR into the Orbiter. It does require the services of the Payload Specialist member of the crew to operate the RMS.

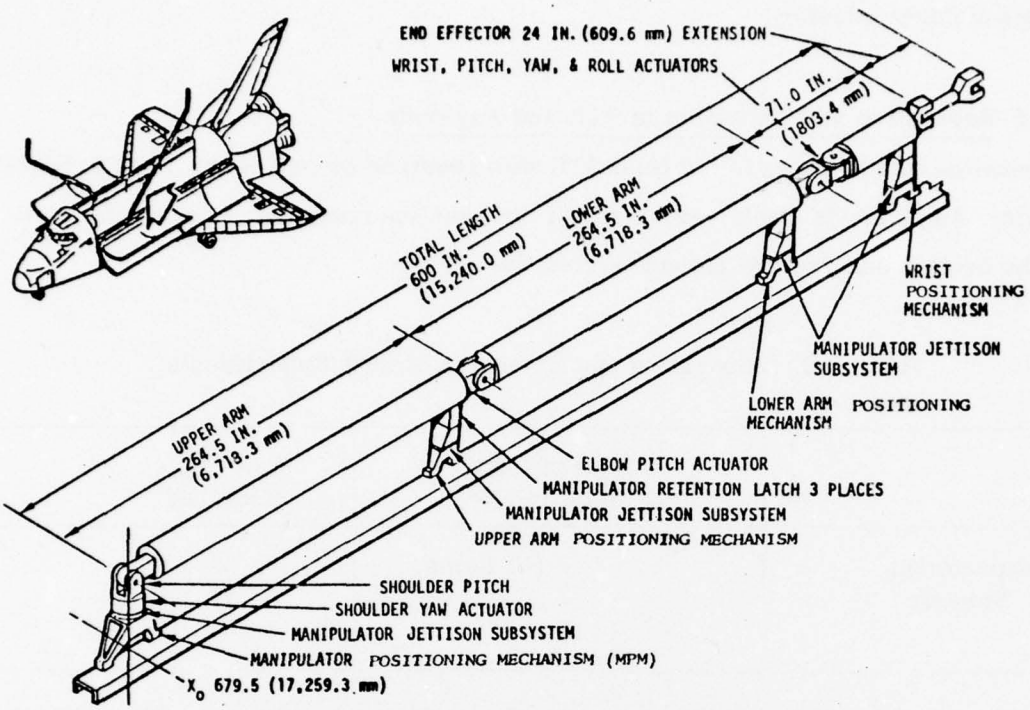


Figure 5-15. Orbiter Payload Deployment and Retrieval Mechanism

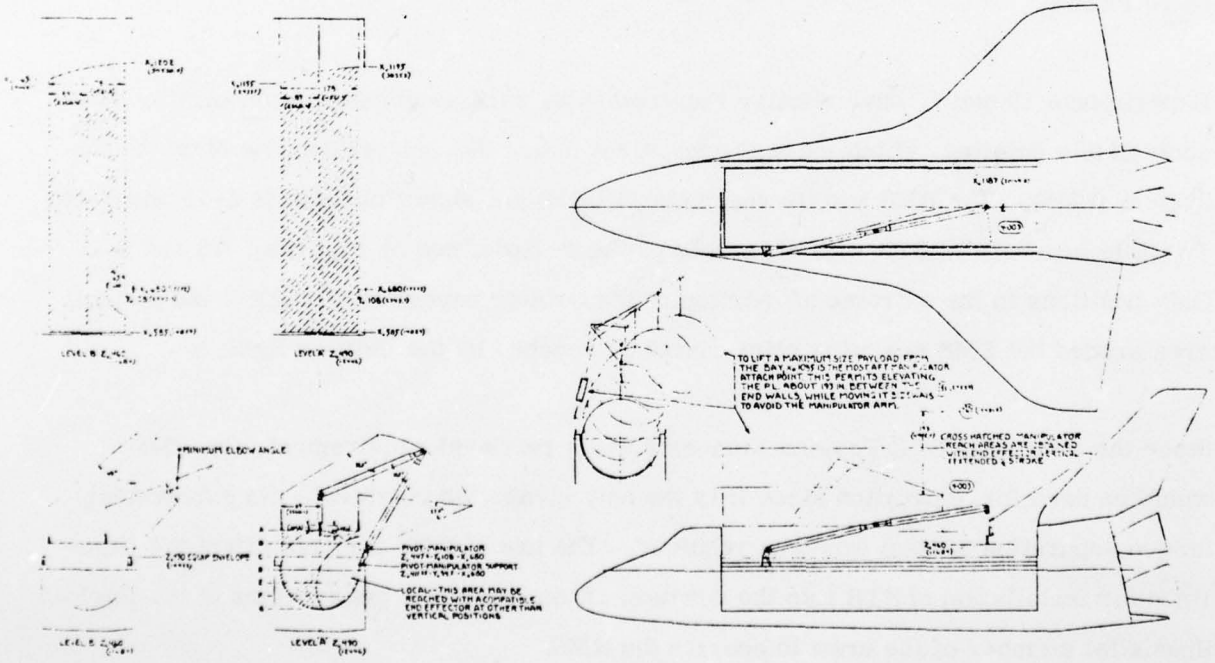


Figure 5-16. Payload Accomodations Master

The remaining baseline experiments and the STR communication subsystem antennas may require extendable booms mounted on the STR to achieve the required field-of-view. This need is a function of the type, size and proximity of other orbiter payloads. The experiment attachment to these booms, the extension and retraction and relatching will all be accomplished by the STR. The interface with these booms and the Orbiter lies in two areas. First, the booms must be jettisonable if the experiment/boom combination protrudes into the orbiter closed-door envelope. Second, a command override capability may be required by the Orbiter crew for safety purposes. Wherever an extendable/retractable boom is utilized, these two interfaces will be satisfied.

5.2.5 CAUTION AND WARNING

Orbiter safety requirements are specified in the following terms:

1. Caution. A hazardous condition which in combination with other system failures may cause loss of the Orbiter or its crew if the condition is not corrected.

2. Warning. A hazardous condition which by itself can cause the loss of the Orbiter or its crew. It is a time critical condition which requires crew action.

Typical situations are premature arming of mechanical or explosive type release and deployment mechanisms, overheating of batteries, failure of pressurized systems elements such as regulators or relief valves.

The Orbiter has a caution and warning system which includes visual and audio alarms, data display and a hazard condition data recorder with replay capability. Prime data from the payloads is fed into the system by dedicated hardwire. Backup data is provided thru the normal payload telemetry interface which inputs to the Orbiter telemetry system. No limits have been defined on the total number of caution and warning signals which the Orbiter can handle.

For the caution and warning system, backup commands to the payload may be provided as part of "crew action" to correct hazardous conditions, if necessary.

A typical payload/Orbiter caution and warning interface is shown in Figure 5-17. Inputs from the STR to the Orbiter C&W system would be similar to the lines marked 5 C&W parameters (hard wire) and 64 Kbps engineering data. The corresponding "crew action" commands are the 5 safing commands (hardwire) and the 36 safing commands.

Current caution and warning studies have been conducted for DOD (Reference 5). The results of these studies indicate that by proper design and use of redundant equipment, the need for C&W signals can be eliminated. This approach is recommended for STR since no C&W would mean a simple interface. Each STR and experiment assembly will have to be reviewed and it is expected that some may require C&W interfaces with the Orbiter. Design requirements should be placed on the payloads so that C&W requirements can be eliminated or minimized.

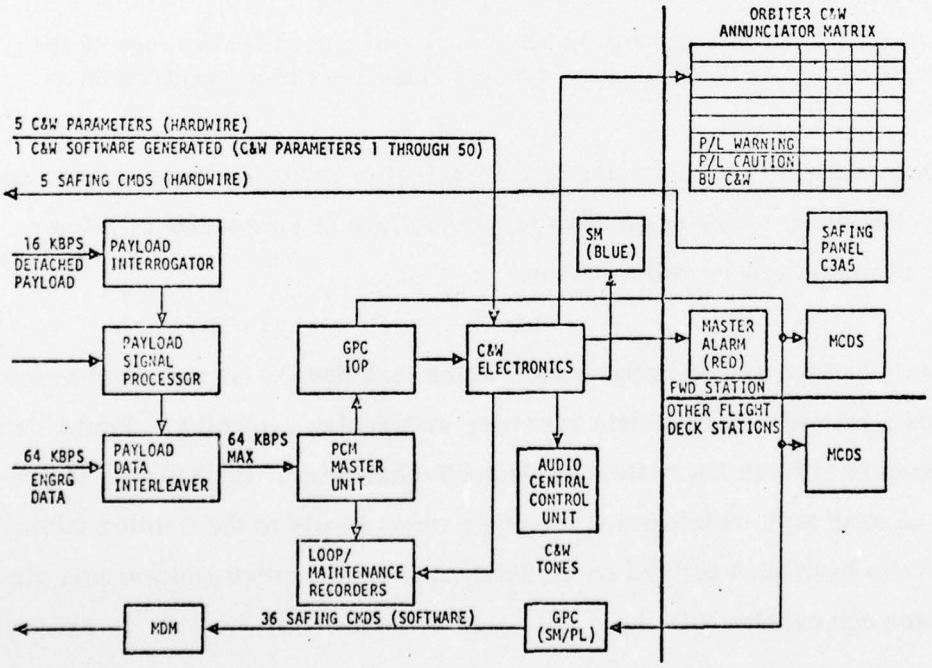


Figure 5-17. Typical C&W Interface

5.2.6 SAFETY

The significant aspect of safety which affects the Orbiter/STR interface is the potential need for command override by the Orbiter crew. They may require some control over the cargo to effect an emergency action not anticipated or readily accommodated by a built in automatic or ground controlled action. This requires an electrical interface and consoles in the payload section of the aft flight deck.

The need and extent of this "requirement" must be negotiated with the STS NASA teams and the Mission Managers. It may be limited to a simple turn-off of the STR system and its experiments. Studies, and contacts as described above, will determine which mode of approach is acceptable. It is expected that some cases will require this interface and that others will be so simple as to make it unnecessary. The interface will be tied into the C&W interface as discussed in paragraph 5.2.5.

5.2.7 ORBITER CREW PARTICIPATION

Participation by the Orbiter crew will be limited to the maximum extent possible to the use of the Payload Specialist for prime payloads. The use of a Payload Specialist will be required only in special cases where complex payloads require it or where the Payload Specialist is scheduled to perform tasks which use up his allowable working time. An alternative to the latter is the use of a second Payload Specialist. A summary of crew participation recommended for baseline experiments is shown in Table 5-10.

Experiments 6, 10, 11 and 12 require use of the RMS and must utilize the Payload Specialist for operation of the RMS. Except for Experiment 7, all other experiments as an alternative could eliminate crew participation and use a totally automated system comprised of the STR computer and prepared software. For R&D one-shot missions, this generally is a costlier approach. The computer would limit hardware costs for this approach by cost sharing over many experiments. The cost of developing and verifying software is the high cost item. It is expected that for simple activities such as turn-on and turn-off, this may be accomplished easily if the time phasing is not subject to limitations of other Orbiter payloads. For cases where pointing, monitoring and adjustment of equipment to acquire data is necessary, crew participation will be

Table 5-10. Crew Participation for Baseline Experiments

Type of Activity	Experiment Number
No crew participation	7
Simple turn-on and turn-off only	3, 9
Monitor of experiment	4, 5
Monitor and Adjust experiments	1, 2, 8
Operate Pointing System	2
Utilization of RMS-Extension & Retraction	6, 10, 11
Deploy experiment - RMS	6, 12
Retrieve Experiment - RMS	6

required. The mission specialist can be utilized for all experiments with the possible exception of Experiments 1, 2 and 8 which will have to be evaluated individually. They may require a payload specialist who is familiar with the experiment. If the Payload Specialist can be trained in less than the time it takes to train a dedicated specialist to be flight qualified (227 hours), it would be advantageous to use the Payload specialist.

A second alternative is to control the experiment operations from the ground station via the SGLS communication system.

5.2.8 FACILITIES AT LAUNCH SITES

The KSC facilities are shown in Figure 5-18. The STR will utilize facilities shown in Table 5-11.

The VAFB facilities are shown in Figure 5-19. The STR will utilize the facilities shown in Table 5-12.

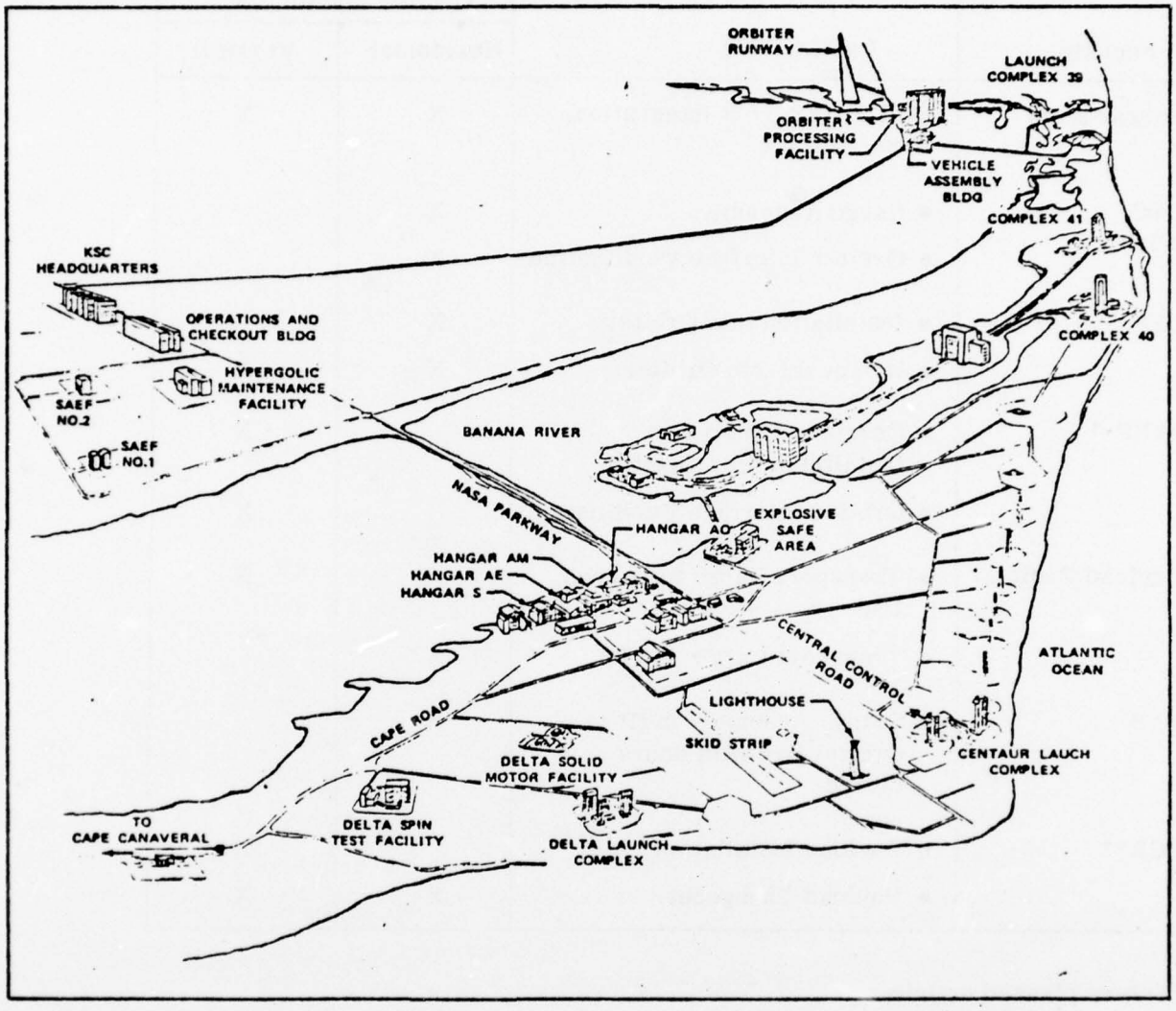


Figure 5-18. KSC Shuttle Facilities

Table 5-11. KSC Facilities

Facility	Use for STR	Orbiter Installation	
		Horizontal	Vertical
Hangar Area	STR/Experiment Integration, Test, Storage	X	X
O&C (CITE)	• Cargo Assembly	X	
	• Orbiter Interface Verification	X	
OPF	• Installation into Orbiter	X	
	• Removal from Orbiter	X	X
SAEF-1	• Cargo Assembly for Installation into Orbiter		X
	• Orbiter Interface Verification*		X
Payload Canister	• Transport Cargo to Launch Pad		X
	• Transport Cargo to OPF	X	
VAB	• Shuttle Assembly (STR in Orbiter bay with doors closed)	X	
PCR**	• Payload Installation		X
	• Payload Changeout	X	X

*Future planned activity

**Not shown in Figure 5-18. PCR is located at launch pad.

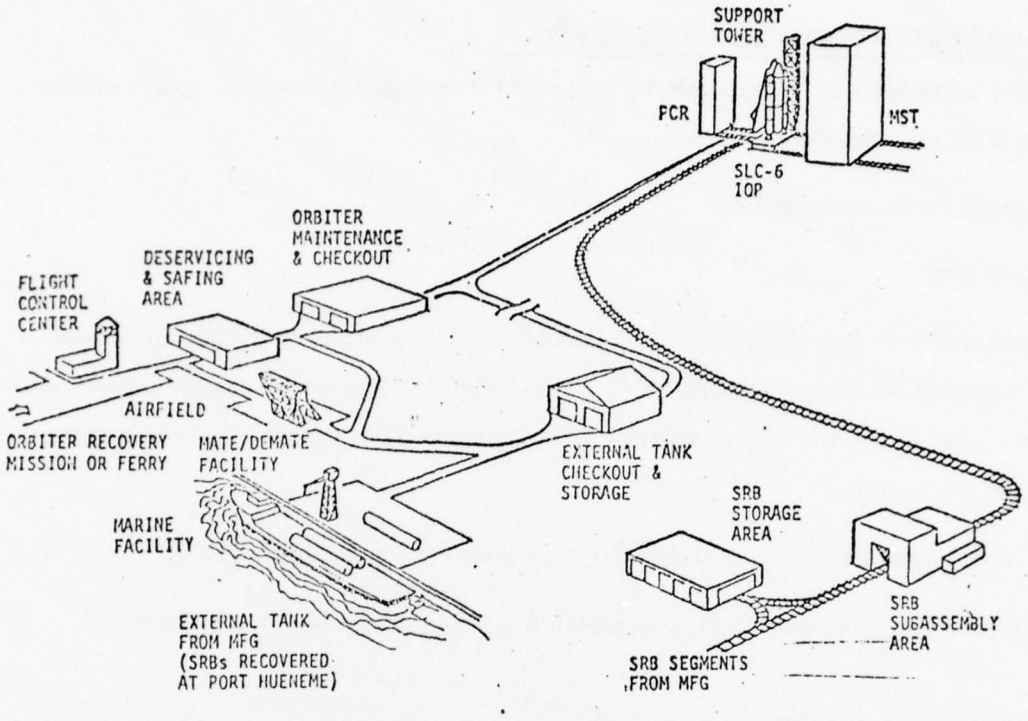


Figure 5-19. VAFB Shuttle Facilities

Table 5-12. VAFB Facilities

Facility	Use for STR	Orbiter Installation	
		Horizontal	Vertical
Satellite Assembly Building*	STR/Experiment Integration, Test, Storage	X	X
PCR (PPR)	Install in Orbiter		X
OMCF	Install in Orbiter	X	
	Remove from Orbiter	X	X
DSA	Post Flight Safing	X	X

*Not defined

Further details of facilities are covered in Section 5.6.

5.3 INTEGRATION APPROACH - STR/ORBITER

Based on the minimal interface achieved by the STR design concept, the goals of integrating the STR into the Orbiter are:

- Quick reaction capability
- Low cost

Quick reaction allows for "last minute" flight of payloads to the maximum possible extent as required by a high priority need or by Orbiter flight space becoming available at a mature point in the pre-flight activity period at any STS flight. There are three types of STR payloads.

1. Experimental equipment being flight proven for operational use
2. Prime mission experiments in which high risk components are qualified
3. Operational missions

The modes of integration of the three types are the same and the following discussion is applicable to all three. The integration approach is to qualify the STR and payloads functionally and experimentally once. The STR is tested to environmental levels (See Section 4.6) with dummy masses enveloping the major configurations. Each payload will be qualified to envelope levels using actual STR mounting panels. Tests are performed at the respective STR and individual payload contractors facility. STR/Orbiter interfaces will be verified once for each STR-payload assembly. This will be accomplished using the proper verification test equipment at KSC and VAFB.

For each STR flight, the STR and payloads will be assembled, aligned and functionally tested to verify payload performance, STR/payload compatibility and interface verification. The compatibility and interface verification of the STR assembly and the Orbiter will be verified insofar as possible using analysis. This covers such areas as structural, thermal, mass and center of gravity, etc. Verification of the physical Orbiter/STR assembly interfaces will be by use of the Cargo Integration Test Equipment (CITE) at KSC and the use of Integration Verfication Equipment (IVE) at VAFB.

(The mobile IVE or the fixed IVE has not yet been selected. For this study the fixed IVE at VAFB is assumed.) Orbiter interface verification testing of STR in CITE or IVE is accomplished with the other Orbiter payloads. This integration approach is summarized in Figure 5-20. A flow diagram representing this approach showing the initial STR flight thru launch and recyle is shown in Figure 5-21. In this flow, all STR integration activities are performed off-line up to Interface Verification. Here the activity is on-line with the Orbiter payload. Orbiter on-line activities begin with the installation of the payload into the Orbiter.

Options for accomplishing the STR/Orbiter integration include the STR flight mode, the Orbiter on-line integration mode, and the STR/payload integration location. These options are listed in Figure 5-22. The details of these options are discussed in the following three sections.

5.3.1 FLIGHT MODES

These are two modes to be considered in STR flight programs. The first is a planned flight mode in which a particular STS flight is identified for a given STR/payload mission in sufficient time prior to launch to be included easily within the normal Orbiter payload integration cycle. This time of identification and committment of the STS flight is anywhere from 4 to 1 years prior to launch. Figure 5-23 shows the activity phases of the planned flight mode integration flow. The early activities consist of the mission planning, integration planning, interface definition and negotiation with STS teams, preparation of key documents such as the Interface Control Document, etc. This activity is done in parallel with development design and testing of the payload and is required to verify flight readiness of the STR-payload assembly. Later activities include the physical integration of payload into the STR, and the physical integration of the STR-payload assembly with the other payloads into the Orbiter. The second flight mode, shown in Figure 5-24, referred to as a flight of opportunity, is one in which the STS mission is identified in the STS prelaunch time period, approximately 1 to 6 months before launch. The latest possible time at which a flight commitment can be made is difficult to predict at this time. It is a function of the payload complexitity and the extent of the interface with the Orbiter. If the payload

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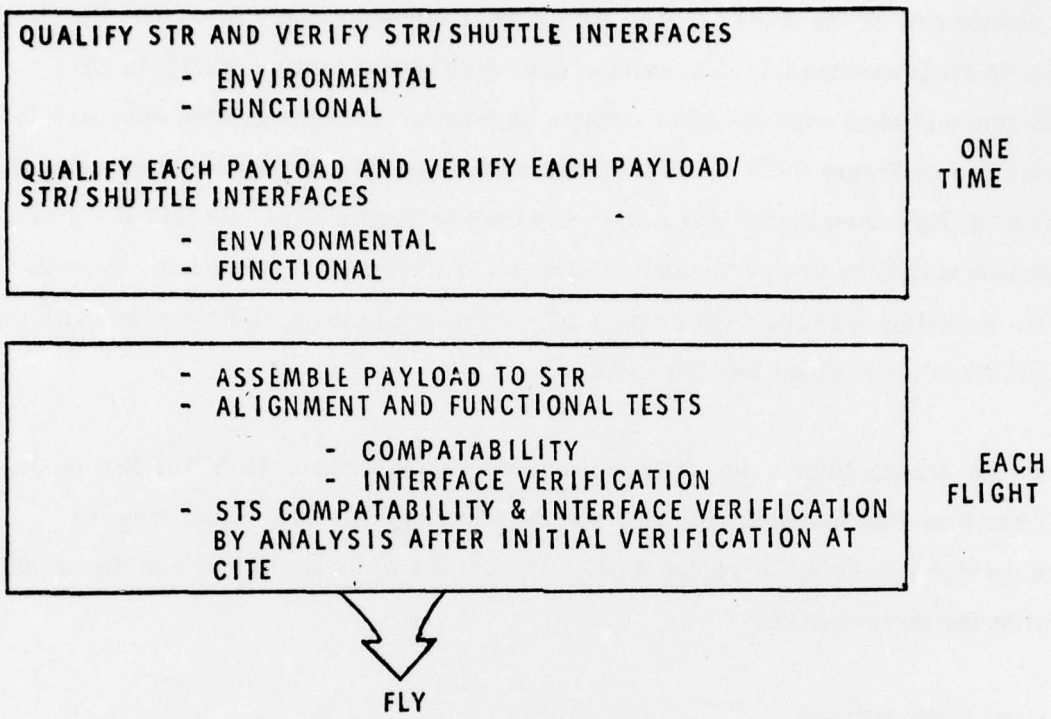


Figure 5-20. Integration Approach For STR

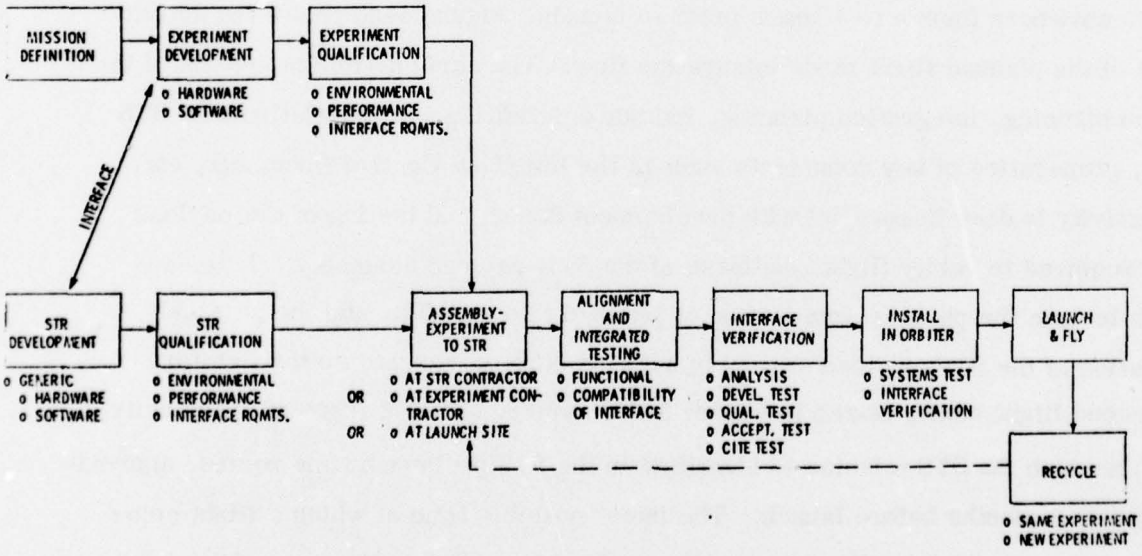


Figure 5-21. STR Integration Flow Diagram

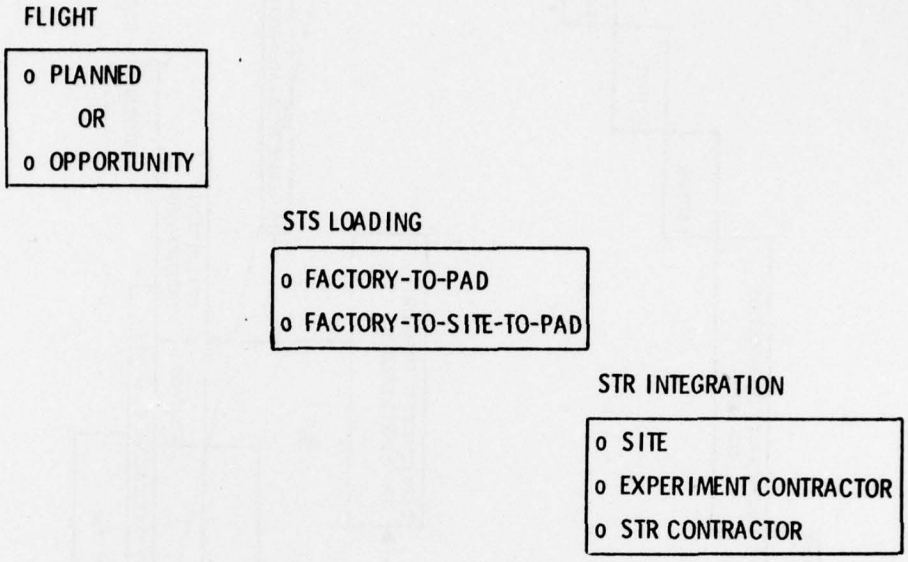


Figure 5-22. STR Flow Options

requires C&W and electrical interfaces or fluid or pneumatic interfaces, this time period can be as long as six months in order to accomplish the integration planning for flight readiness verification. A shorter time period is required to accomplish the physical integration phase. Use of unused Orbiter facilities such as C&W and command capability, STR console room on the aft flight deck, etc. must be planned for and accomodated by NASA not only for STR but for any "last minute" payload.

Much of the flight verification effort such as the structural, electrical, thermal, safety, C&W, mass properties etc. can be done generically prior to a flight commitment. However providing STR data and analyzing total payload must be accomplished by a prime payload integrator or mission manager. It is this activity which requires 1 to 6 months prior to launch. Once the planning and verification activities are complete, the physical integration for both modes is carried out in an identical fashion.

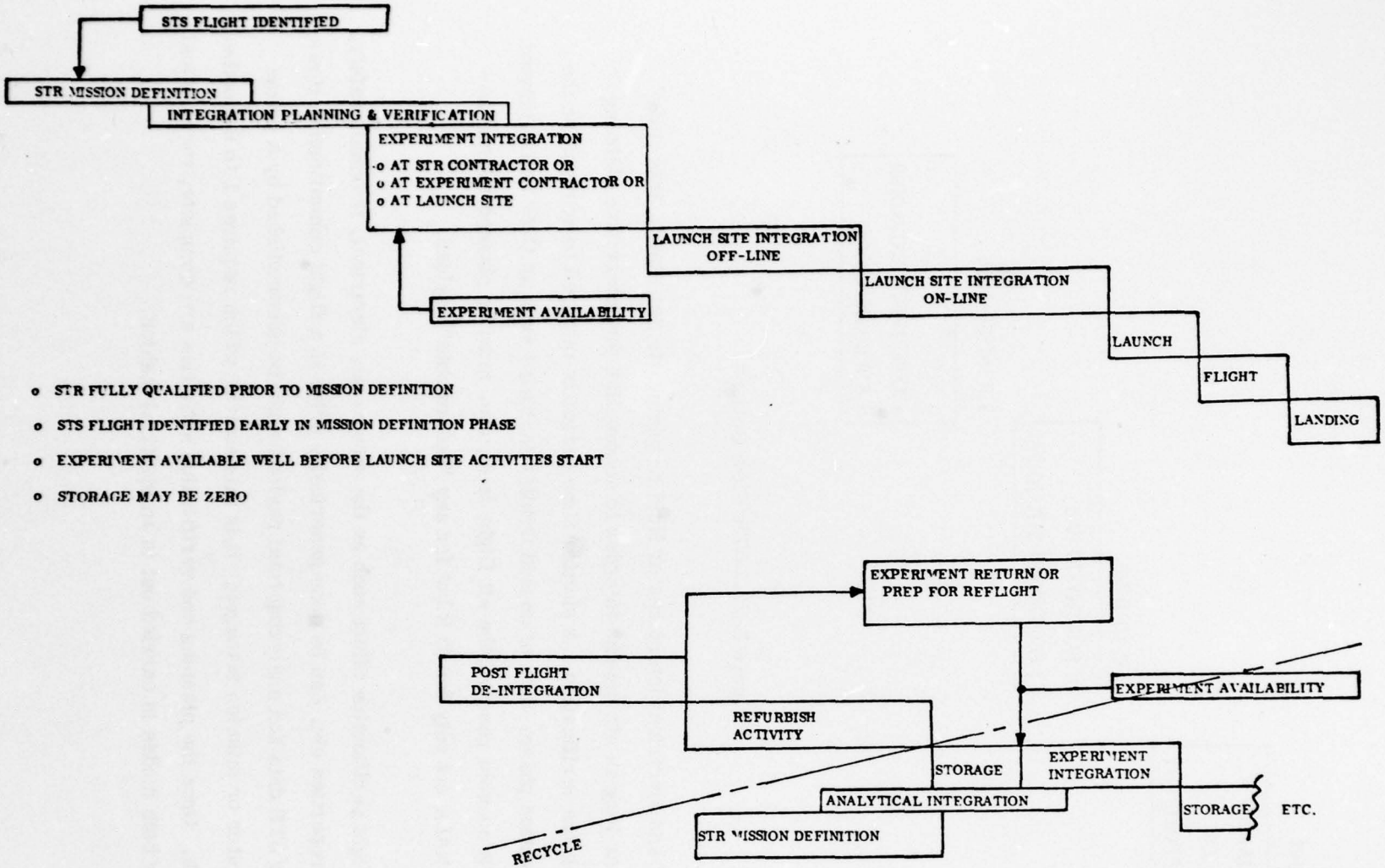


Figure 5-23. Planned Flight Mode

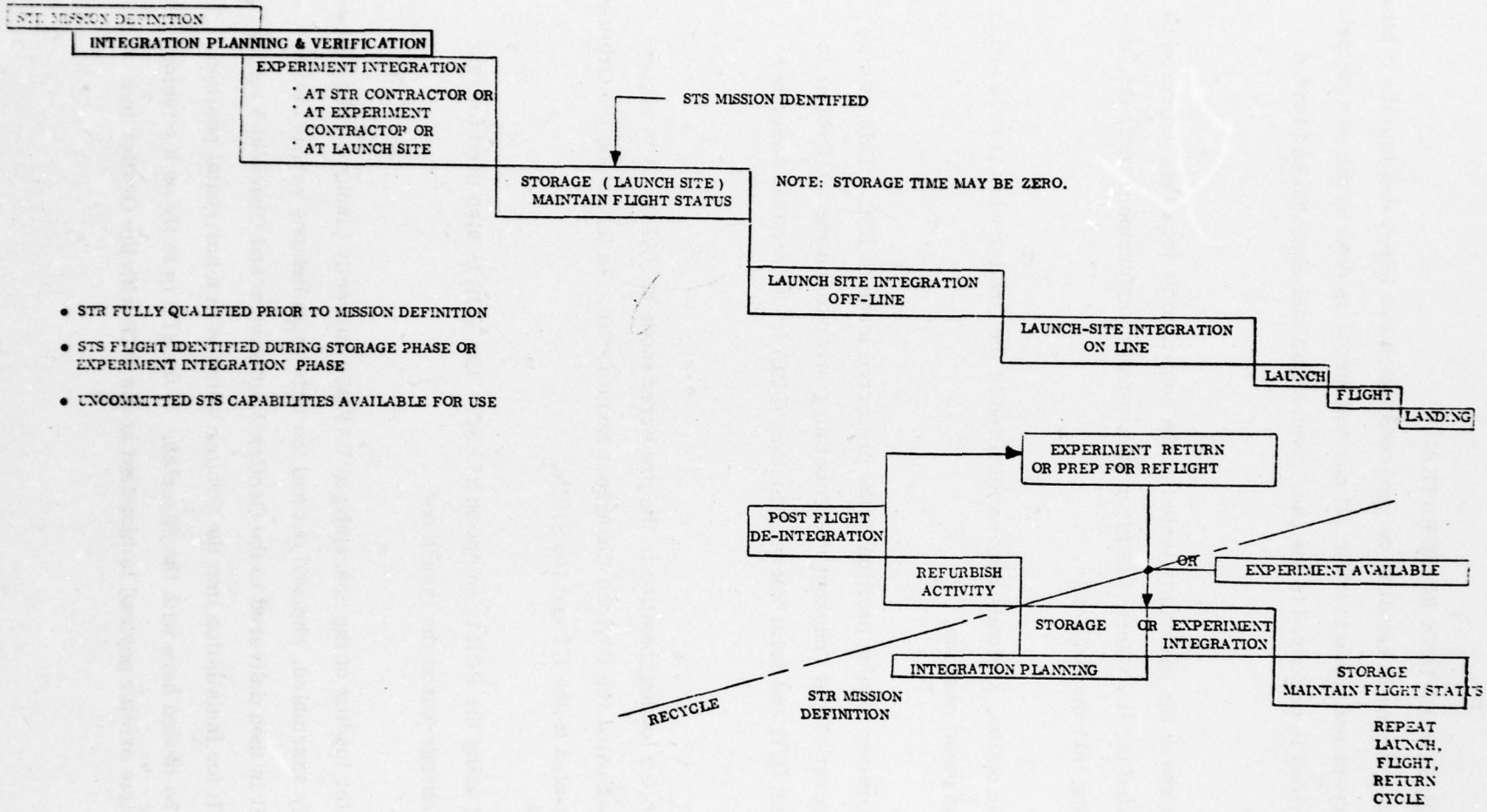


Figure 5-24. Flight of Opportunity Mode

5.3.2 ORBITER/STR ON-LINE INTEGRATION

There are two approaches to handling the payloads which are integrated into the orbiter. One is the factory-to-pad mode in which all payload testing is done at the contractor's location, the payload is delivered to the pad, given a minimal checkout and loaded into the Orbiter.

In the second approach the payload is tested at the contractor's location, delivered to a payload dedicated facility where it is given an extensive performance test, then delivered for loading into the Orbiter.

In addition to this option, payloads may be installed in the Orbiter when it is in either a vertical or horizontal position.

The horizontal Orbiter loading position is the preferred mode at KSC. This loading occurs in the Operation and Checkout (O&C) building prior to mating the Orbiter to the External Tank (ET) and Solid Rocket Boosters (SRB) in the vertical Assembly Building.

The vertical Orbiter loading position is the preferred mode at VAFB. It is accomplished on the pad using the Payload Changeout Room (PCR). In this mode the Orbiter is already assembled to the ET and the SRB's.

Vertical loading using the PCR is an option at KSC. The PCR is also used for last minute payload change-out at the launch pad.

Horizontal Orbiter loading of the Spacelabs at VAFB is currently planned. The Spacelab is completely assembled, checked out and the Orbiter interfaces verified in the CITE at KSC. It is then delivered to the Orbiter Maintenance and Checkout Facility (OMCF) at VAFB for installation into the Orbiter which is in a horizontal position. The STR would be loaded here with the Spacelab. If the STR is to fly on a mission on which the prime orbiter payload is installed at the PCR with the Orbiter in a

vertical position on the pad, then the STR would be installed at the PCR. Only one mode of loading would be used for any STS flight with STRs to keep down the integration costs. The integration flows for STR with Spacelab and non-Spacelab payloads at KSC and VAFB are shown in Figures 5-25 through 5-28.

STR flow without Spacelab at KSC utilizes a canister (part of STS ground support equipment) which duplicates the payload bay volume and mounting interface. Orbiter cargos are assembled in the O&C building, installed in the canister and moved to the PCR. An alternative is to deliver payloads separately by canister from their assembly points, such as SAEF-1 for an IUS-Spacecraft combination, to the PCR. The PCR payloads are transferred to the Payload Ground Handling Mechanism (PGHM), and from there to the Orbiter bay.

At VAFB, all payloads (except Spacelab) are delivered to the PPR, then to the PCR, mounted on the PGHM and then installed in the Orbiter bay. The STR can be assembled

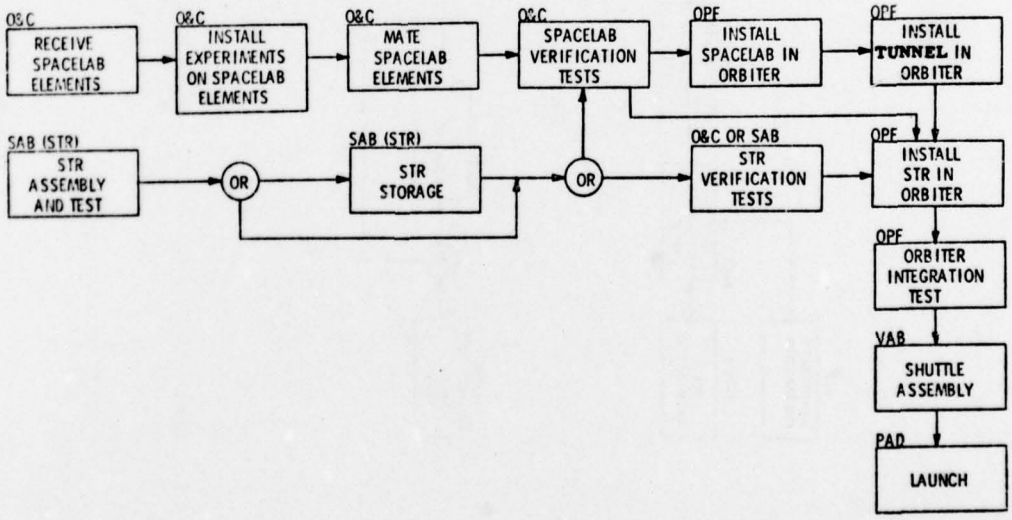


Figure 5-25. STR Flow with Spacelab KSC

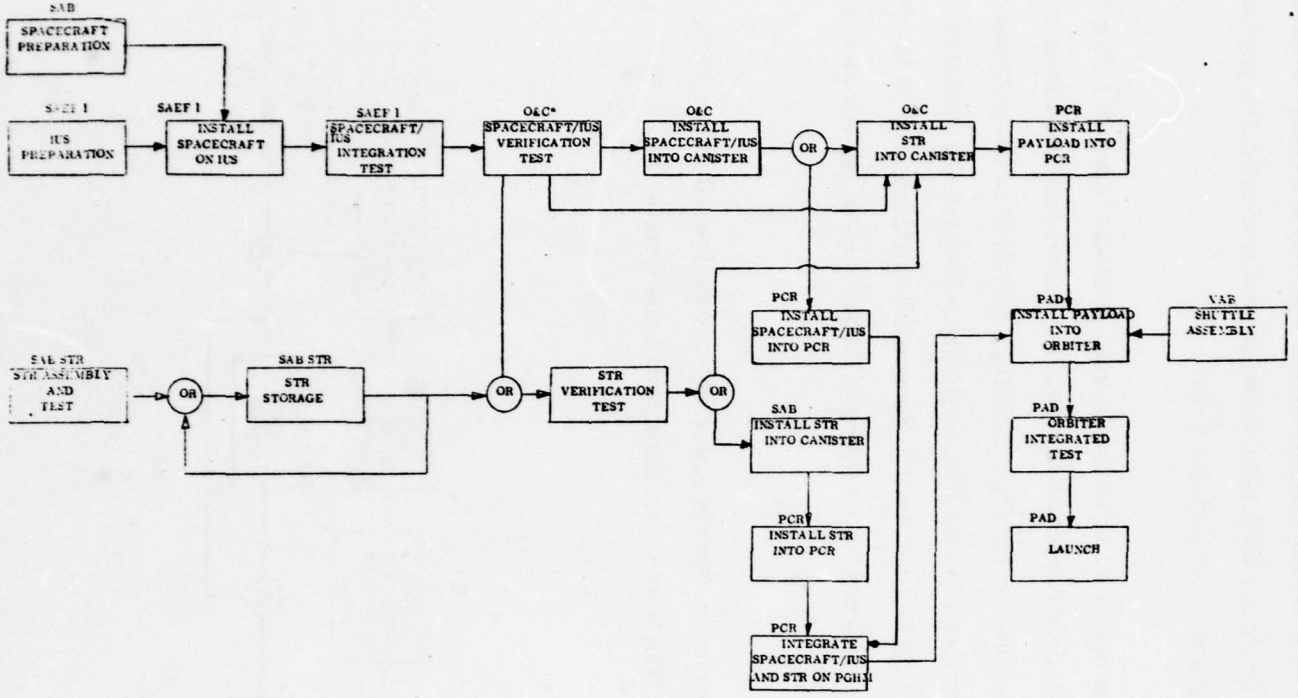


Figure 5-26. STR Flow with IUS/Spacecraft KSC

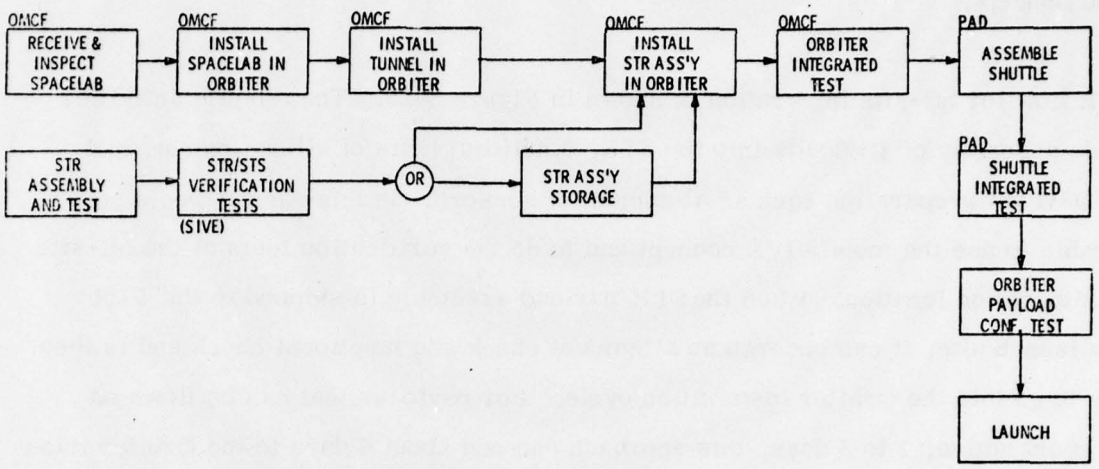


Figure 5-27. STR Flow with Spacelab - VAFB

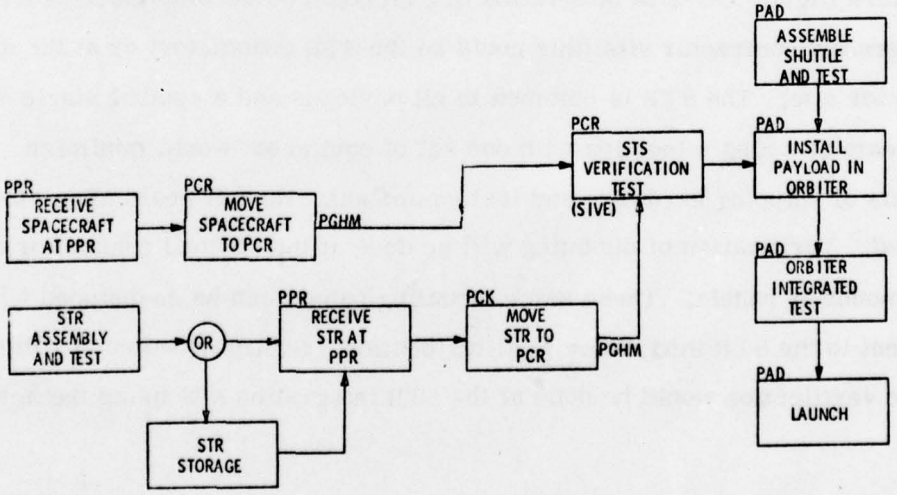


Figure 5-28. STR Flow with Spacecraft/IUS - VAFB

and tested in preparation for integration into the Orbiter on or off the launch site. The off-site assembly and test of the STR and payloads lends itself to the factory-to-pad concept.

A STR flow for off-site integration is shown in Figure 5-29. The off-site activities include assembly of payloads into the STR, functional tests of all equipment, and STR/payload preparation such as alignment of sensors. In this case it would be desirable to use the mobile IVE concept and to do the verification tests at the off-site STR integration location. When the STR/payload assembly is shipped to the PPR at the launch site, it can undergo an alignment check and functional check and is then ready to go into the Orbiter installation cycle. For payloads that can be flown on very short notice, 1 to 5 days, this approach can add about 4 days to the Orbiter integration cycle to accommodate packing, shipping by local truck and air, moving, and unpacking the STR. This approach needs no facility space outside the PPR and PCR for the STR. A modification of this mode would be to complete all the aforementioned activities off-site, but to ship the STR to the launch site and to store it in a dedicated facility. Storage would be in a flight ready condition. This would minimize the reaction time before flight. Off-site integration of STR could be accomplished at a selected payload integrating contractor site (this could be the STR contractor) or at the experiment contractor site. The STR is common to all payloads and a central single integrating team at a single location with one set of equipment would minimize logistics costs of shipping hardware and test equipment. No full scale simulators are recommended. Verification of mounting will be done at the payload contractor using actual STR mounting panels. These same mounting panels can be co-shipped with the experiment to the STR integration facility/location. STR/payload compatibility and interface verification would be done at the STR integration site using the actual hardware.

An on-site integration flow is shown in Figure 5-30. In this mode a dedicated on-site facility area of about 7000 square feet would be required. (see paragraph 5.6). All STR and experiment assembly, test, and storage would be accomplished at this site.

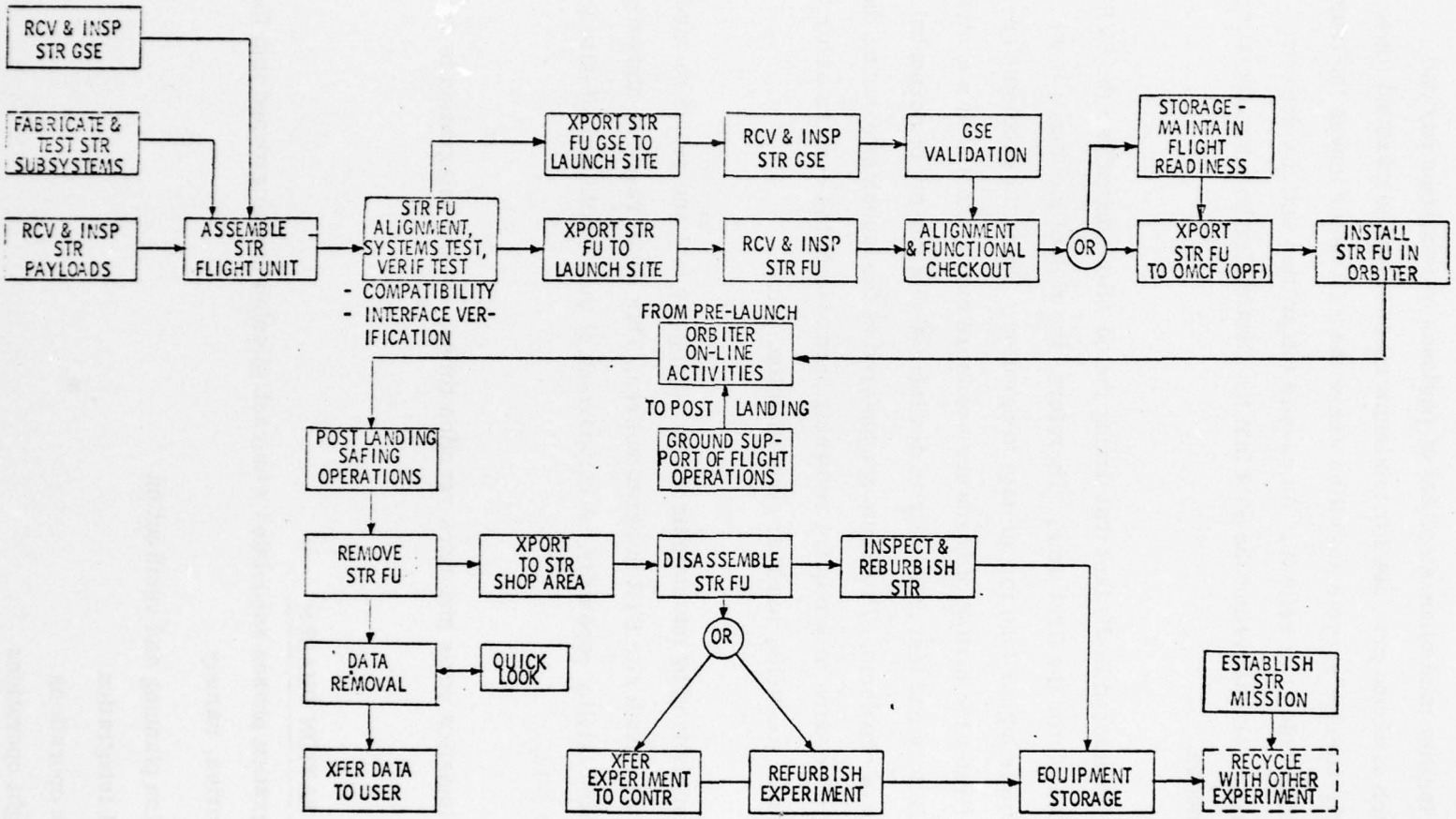


Figure 5-29. STR Operational Flow Diagram Off Site Integration

Use of an on-site facility minimizes shipment of payloads (direct from payload contractor to launch site) and provides for minimum reaction and turnaround time. This is particularly true for recycle activities where the STR never leaves the launch site and for payloads which are reflown. An on-site integration activity does not change the timelines for integrating the STR into the Orbiter for both horizontal and vertical loading modes.

A last item to be considered is the fact that during the on-site integration, the STR and experiments are mated for the first time. Therefore, the risk of meeting a "last minute" launch date is higher than for off-site integration. Potential problems encountered during first-time mating of hardware are more easily handled at a contractor site where a full factory and test capability is available with a variety of personnel to handle many types of problems. Problems encountered at the launch site during the first-time mating procedure may require returning hardware to the contractor for test to determine problem solutions, hardware modification, etc.

This risk can be minimized by maintaining a more complete test and repair or modification facility at the launch site STR integration area. This could result in higher costs to support this added facility capability. A comparison of the on-site and off-site location is shown in Table 5-13.

To minimize the logistics costs and quick reaction times, on-site integration is recommended.

5.4 STR INTEGRATION PHASES

The various integration phases associated with each mission can be grouped into the four major categories, namely

1. Integration planning and verification
2. Physical integration
3. On orbit operations
4. Post flight operations

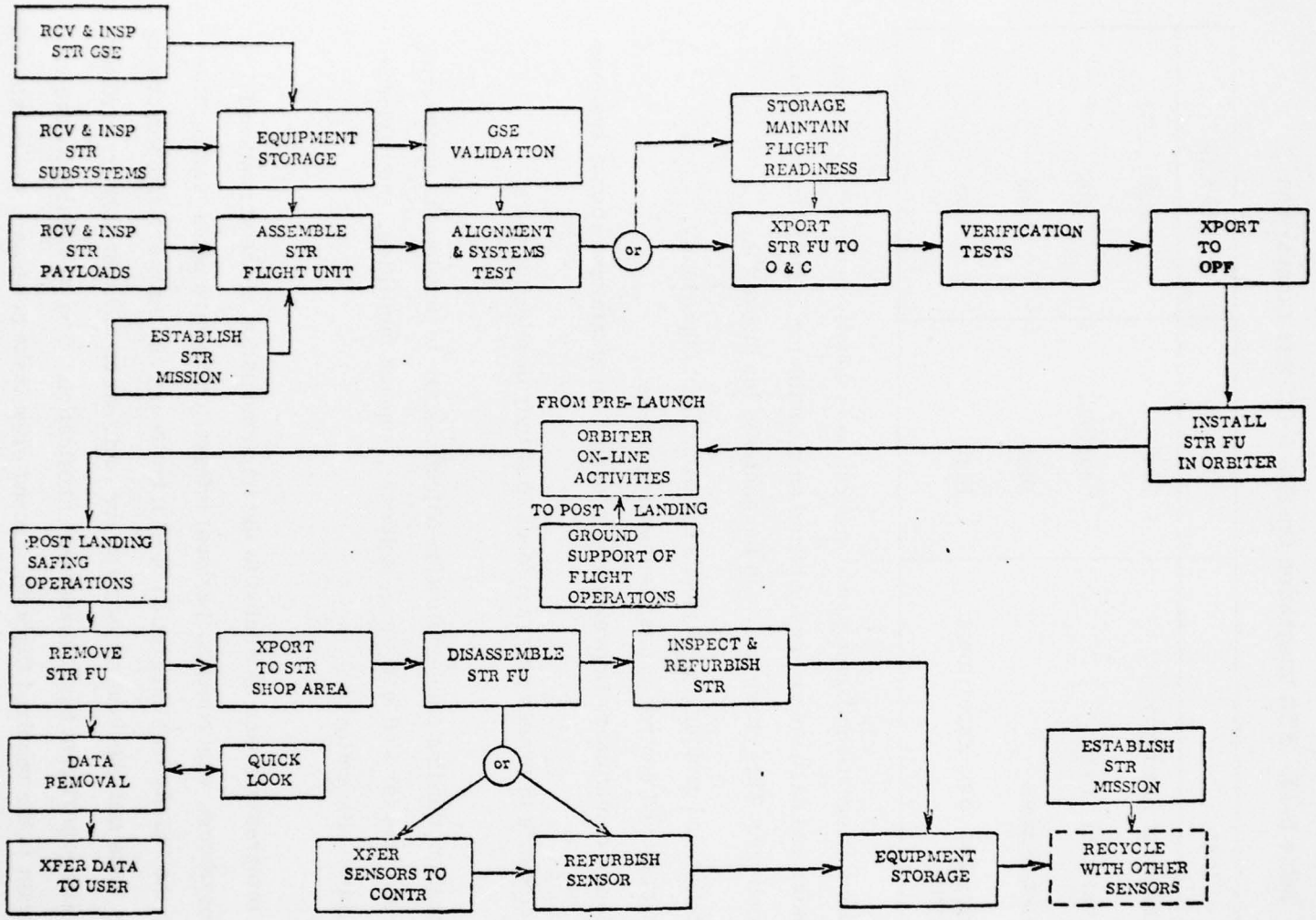


Figure 5-30. STR Operational Flow Diagram - On-Site

Table 5-13. STR Integration On-Site & Off-Site Comparison

Item	On-Site	Off-Site
• Shipment of Hardware	Low	High
• Turnaround Time	Shortest	Long
• Facility Space	High	Low
• Integration Risk-Experiment to STR	High	Low

The initial phase integration planning and verification is usually related to payloads/ mission planning and analysis and is conducted on a continuing basis to meet approved program objectives. This phase provides the analyses leading to the:

1. Selection and grouping of payloads for each flight opportunity,
2. Allocation and control of total project resources,
3. Effective evolutionary improvement of overall program operations, systems and facilities,
4. Maintenance of proper program focus to satisfy developing goals.

Typical integration planning and verification objectives are to develop the engineering and operational plans for STR hardware, software, support faciliities, and payload-to-STR-to-orbiter integration.

The physical integration phase is related to the implementation of the various (STR structure/subsystems, experiments, payload) integration requirements plans. This would include the "hands-on" integration of STR hardware and software and performing the necessary tests and checkout procedures for verification of flight support hardware performance and experiment flight readiness certification. The on-orbit integration activities pertain to the required flight and ground crew tasks necessary to operate and monitor the payloads. The requirements for these activities are developed during the integration planning and verification phase, procedures specified within the flight crew activity, and payload operations control center operations and support plans.

The post-flight operations phase includes landing site operations and post-mission support and logistics of STR hardware and experiments. These tasks would provide for disposition of experiments and expendables, where applicable, data collection and processing, and payload/STR receipt, inspection, and post-flight tests and checkout, and also handling of special equipment/GSE. Each of these areas is discussed more fully within the following text.

5.4.1 STR INTEGRATION PLANNING AND VERIFICATION

This integration phase provides the analyses and plans for mission definition and integration as indicated in Figure 5-31. On a mission-to-mission basis, this activity is the most extensive as to manhour requirements and overall time period to accomplish.

The mission definition phase of the STR integration planning activity is directed towards tracking the STR mission model, cargo manifests, schedules, and flight assignments. The purposes of this activity is to provide the baseline information on specific mission objectives and payload configuration data, i.e., proposed experiments, mass properties,

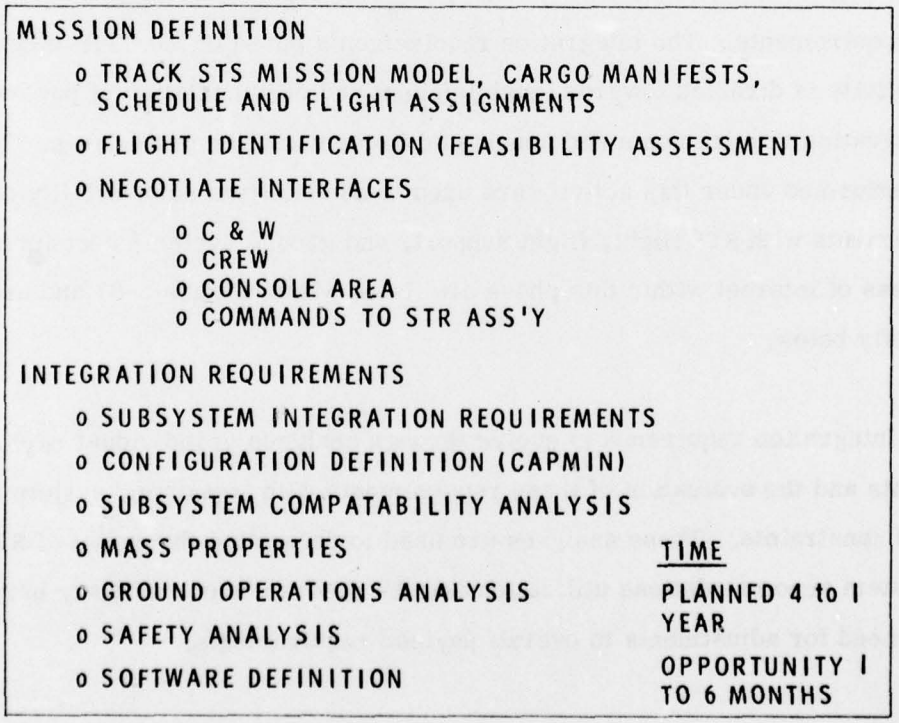


Figure 5-31. Analytical Integration

and resource requirements. This baseline information is obtained from Integrated Payload Requirements documentation which is prepared for each mission by the responsible Payload Integration Contractor/NASA team. The STR integration planning team will use these top level documents in order to determine applicable STR/mission payloads. More detailed analyses relating to capability and resource requirements; i.e., data/command/communications, environmental control, electrical/structural/mechanical, pointing/stability, etc., can be found in the Integrated Payload Compatibility Analysis Summary and associated referenced documentation for each mission. These payload compatibility analyses will provide the data against which STR/experiment compatibility (feasibility) assessments can be made for identification of the appropriate flight for STR utilization.

After an STR/flight combination has been identified, the STR analytical team will be required to review the Interface Agreements for that mission in order to negotiate the required interfaces. These will include physical connections, added requirements for safety, and resource allocations. Specific areas to be considered for the STR package are: caution and warning, crew involvement, console area changes, and functional command requirements. The integration requirements phase of the STR integration planning activity is directed towards establishment and documentation of payload/ground integration requirements and associated implementation procedures. The analyses performed under this activity are used to demonstrate compatibility of the STR/experiments with STS flight, flight support, and ground systems accommodations. Typical areas of interest within this phase are listed within Figure 5-31 and are discussed briefly below.

Subsystem integration requirements evolve through analyses of individual payload requirements and the evaluation of these requirements with supporting system capabilities and constraints. These analyses are used to determine the extent of STR/Orbiter system accommodations utilization and the need for supplementary hardware and/or the need for adjustments to overall payload requirements.

Configuration definition provides the design analyses necessary to define the STR payload configuration layout, develop installation/assembly definitions, perform STR peculiar hardware design, and develop end item specifications. Low cost configuration studies can be performed using the GE developed CAPMIN (Computer Aided Payload Mission Integration) program. This program will not only provide a fully automated procedure for configuration/design documentation but also provides the capability to implement a word processing system for reducing integration requirements and procedures documentation costs.

Subsystem compatibility analyses are used to demonstrate compatibility of payload to STR integration as pertaining to command/data/communications, environmental control, electrical, structural/mechanical, and pointing/stabilization requirements. In addition, analyses are performed to assure compatibility among payload system elements during the ground, ascent, on-orbit, descent and post landing mission phases.

Mass properties are developed and maintained from initial flight planning to flight to reflect the current STR/payload configuration. STR/payload compatibility assessments with mission requirements are performed in order to assure mission success.

Ground cperations analyses are performed to define the requirements for integration, verification, and support, and to insure the provisioning of GSE, facilities, and software to meet these requirements. Also included are the development, preparation, and maintenance of the physical and functional requirements documentation for the implementation of physical integration of the STR payloads.

Safety analyses are performed to provide the definition of requirements associated with identification and control of potential safety hazards that must be considered in the design, development, fabrication, test verification and safety compliance data of the integrated STR payload. Experiment/instrument developers are responsible for hazard identification, control and assessment of their devices. The STR/payload integration, launch, site, flight and post-flight flows are analyzed and identified

hazardous conditions are resolved by design, analysis, procedural, inspection, and/or operational work arounds in accordance with NHB 5300.4/1D-1) paragraph 1D203.

Software definition must be provided to meet ground processing and flight operations requirements. Ground processing software is that software which is required during STR integration to support experiment hardware and software integration, checkout, and validation. Flight applications software is used to control the experiment operation during flight as well as for ground checkout when executable in a one-g/one-atmosphere environment. A simulated flight program may be necessary when the experiment will not operate during ground operations in order to provide for the STR Integrated Systems Test.

Depending on the complexity of the STR payload and the extent of manpower applied to the integration planning activity, this analysis phase must start anywhere from one to four years prior to launch for planned flights. These numbers are based on current assessments related to early missions. For flights of opportunity where a fully flight certified and integrated STR payload is in storage, it is expected that this activity must be started anywhere from one to six months prior to launch. This time period would be required to analytically checkout and verify Orbiter/STR interfaces, verify compatibility of the STR payload with the designated primary mission payload, and issue NASA/DOD approved documentation necessary for the physical implementation of the STR payload into the Orbiter. With simple or fully automated payloads requiring little interfacing with the Orbiter (e.g. no caution or warning or crew involvement), or as Shuttle payloads are reflown and maturity is increased, the extent of compatibility/interface verification can be significantly reduced such that the time duration prior to launch may be as little as one month (primarily for preparation and issuance of the NASA/DOD documentation).

5.4.2 STR HANDS-ON PHYSICAL INTEGRATION

A top level "building block" approach to the "hands-on" (physical) integration ground processing activities is shown in Figure 5-32. The Figure indicates the various

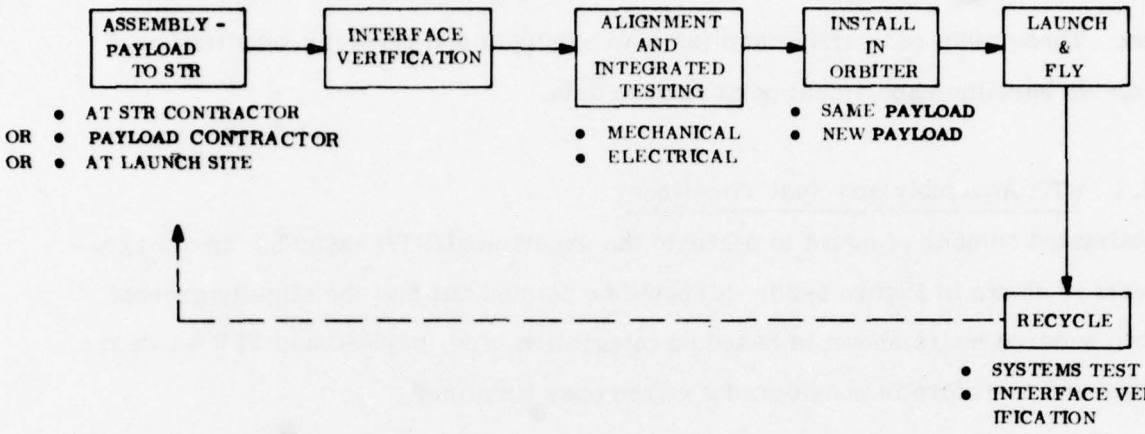


Figure 5-32. Physical Integration

"hands-on" activities associated with each phase of the ground processing cycle from initial STR/payload assembly to recycle of the STR assembly after flight. Although three possible locations are specified at which the STR/payloads can be assembled, it is recommended that the launch site be selected to minimize the turnaround time cycle by eliminating the added activities associated with packaging and shipping flight articles to other locations (and also to avoid possible damage due to shipment and extra handling procedures).

Various integration tests are required to assure that the flight unit is indeed compatibility compliant and flight worthy. The tests are primarily mechanical and electrical in nature and are performed in a logical sequence with hardware build-up.

Mechanical integration tests pertain to mounting/fit, alignment, and accessibility. This would entail installation of experiments and subsystem elements into the STR, validation of mechanical interfaces, verifying alignment of critical components, and validating crew accessibility (if required).

Electrical integration tests pertain to payload/subsystem compatibility checks, software checkout, and functional performance verification.

Systems tests are performed to verify electromechanical compatibility (EMC) of the fully assembled flight configuration and to ferret out RF interferences and susceptibilities. These tests can verify downlink power outputs and receiver sensitivities, and provide baseline experiment performance data.

5.4.2.1 STR Assembly and Test Timelines

The estimated number of hours to perform the experiment/STR assembly and integration tests is shown in Figure 5-33. It should be pointed out that the slightly greater than two hundred hours shown is based on integration of all payload and STR hardware elements and therefore is considered a worse case timeline.

Although the STR will have to be assembled/disassembled for each mission, not all components will have to be removed or all interfaces verified each time. Therefore it is realistic to assume that the timeline shown in Figure 5-33 can be reduced as systems are reflown. The objectives of this integration activity, as Shuttle maturity increases, will be to assure that the STR turnaround time be always less than the turnaround times assessed for the Orbiter. Current estimates for the Orbiter turnaround times are greatly in excess of two hundred hours, with many early mission-to-mission configuration combinations having times as high as four hundred hours. It is estimated that the STR turnaround time cycle could, as systems are reflown, be improved to meet the Orbiter allocated (optimistic) turnaround time cycle of one hundred sixty hours. The fact that Orbiter assessed turnaround times are greater than one hundred sixty hours provides more than adequate time for the STR refurbishment/reconfiguration activity to be accomplished.

The first phase of the timeline activity shown in Figure 5-33 is applicable to the build-up of the STR structures and subsystems and will be less for reflown units. The second

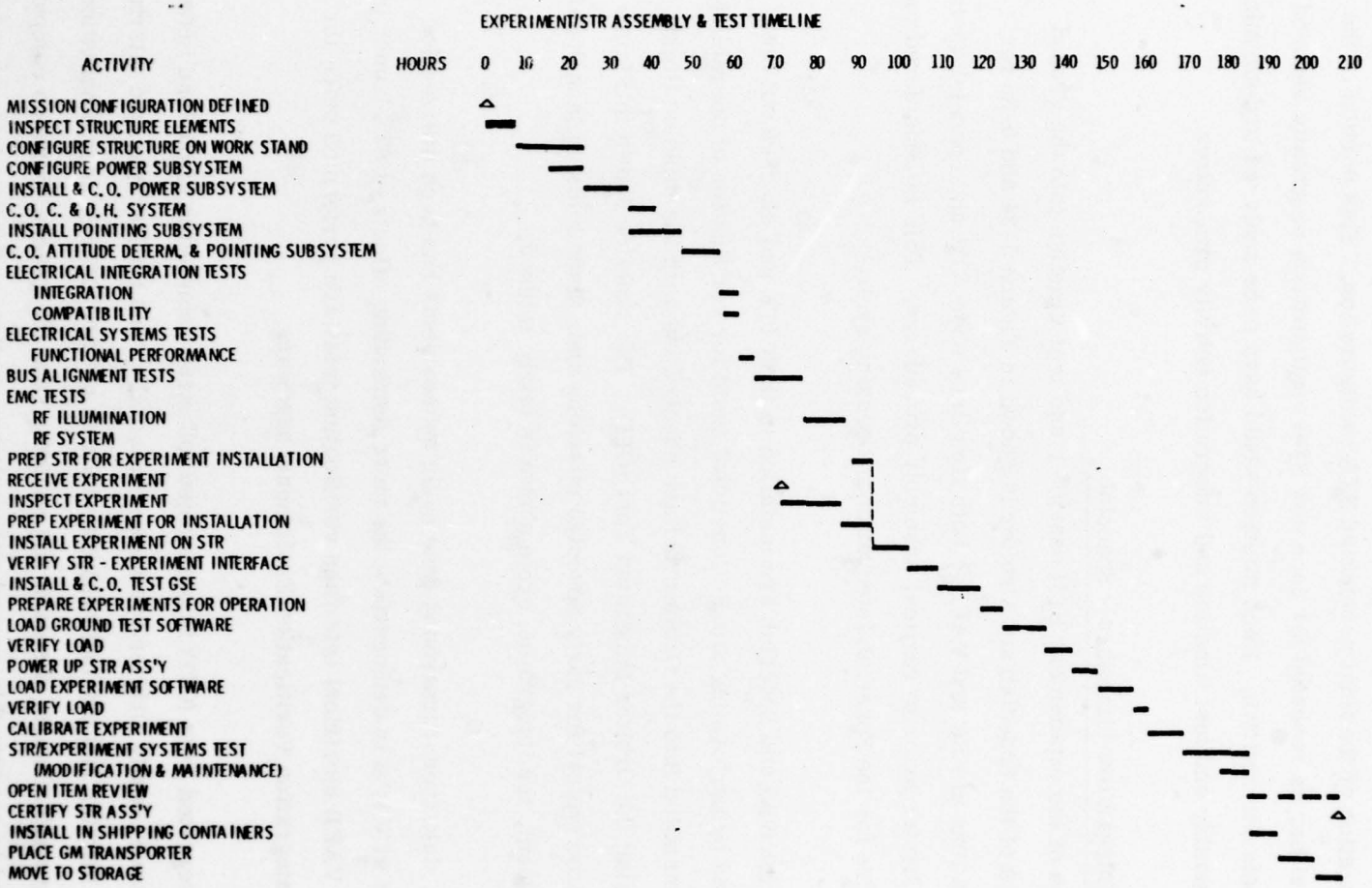


Figure 5-33. Experiment/STR Assembly & Test Timeline

phase of the timeline activity is related to installation, integration and functional performance verification of the final assembled STR configuration. This portion of the timeline would either be repeated for each new STR configuration or greatly reduced if the experiments are reflown. Each mission would have to be analyzed to determine the specific assembly and test timeline and integration activity procedures.

5.4.2.2 STR Integration Timelines - Spacelab

The introduction of the experiment/STR assembly and test timeline into the ground processing cycle of the Spacelab and Orbiter is shown in Figure 5-34 and 5-35 for prelaunch operations at KSC and VAFB. Both timelines show STR and Spacelab installation into the Orbiter prior to vertical assembly procedures. This is considered the normal procedure for the Spacelab integration processing cycle.

In Figure 5-34 the STR and Spacelab are installed in the CITE and checked out both mechanically and by performing simple functional tests for verification of operability. They are then installed into the Orbiter and are checked out with the mission flight subsystems during the Orbiter Integrated Test (OIT). The timeline shown in Figure 5-34 is considered typical for early Spacelab missions and, since mission launch dates are not too frequent, the integration cycle shown is fairly relaxed.

In Figure 5-35, data from turnaround time cycle assessments has been utilized for STR integration at VAFB to demonstrate the more demanding effects of STS launch pad integration. At VAFB electrical interface verification tests are performed using the SIVE (Shuttle Integration Verification Equipment) hardware.

The Orbiter Integrated Test (OIT) is performed following completion of payload installation. This test will verify Orbiter systems compatibility and Orbiter/payload interfaces. The necessary Orbiter and payload systems (such as power, ECLSS, instrumentation, data processing and software, displays and controls, and the general purpose computer) will be activated to support this ground processing operation. Unique mission/payload accommodation kits will be checked and verified, i.e., the payload-peculiar control

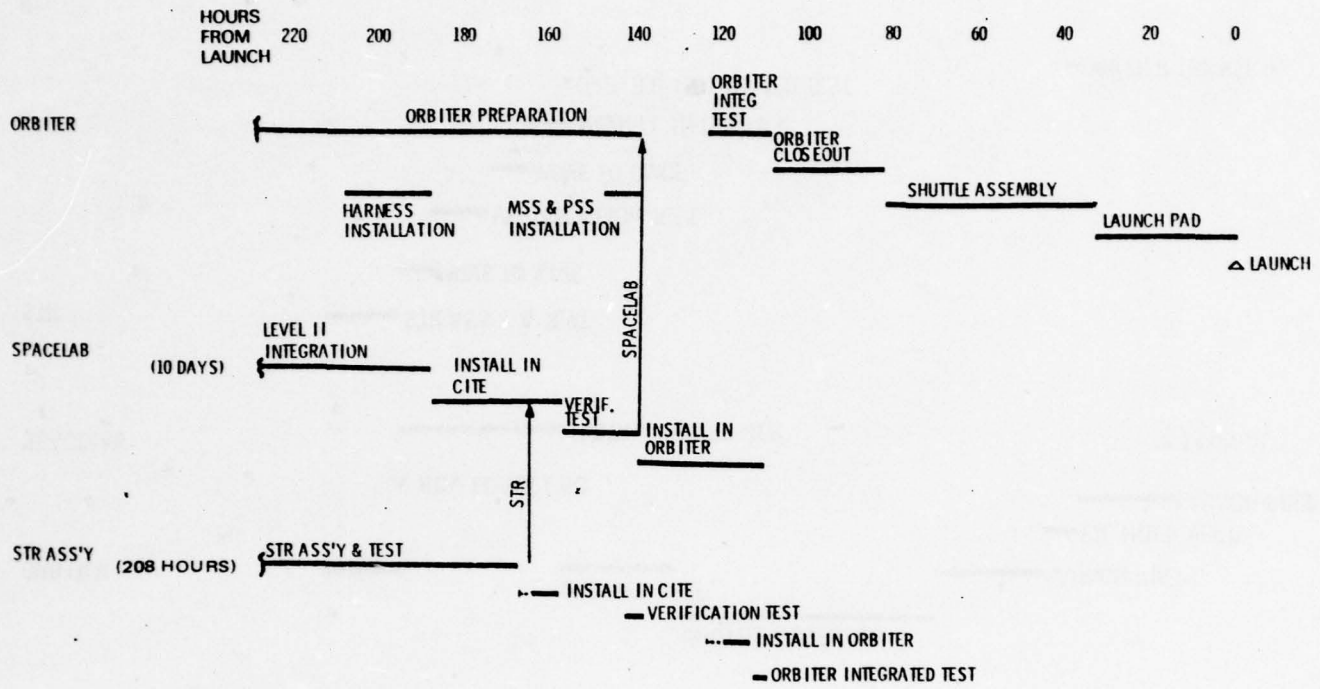


Figure 5-34. STR Timeline with Spacelab - KSC

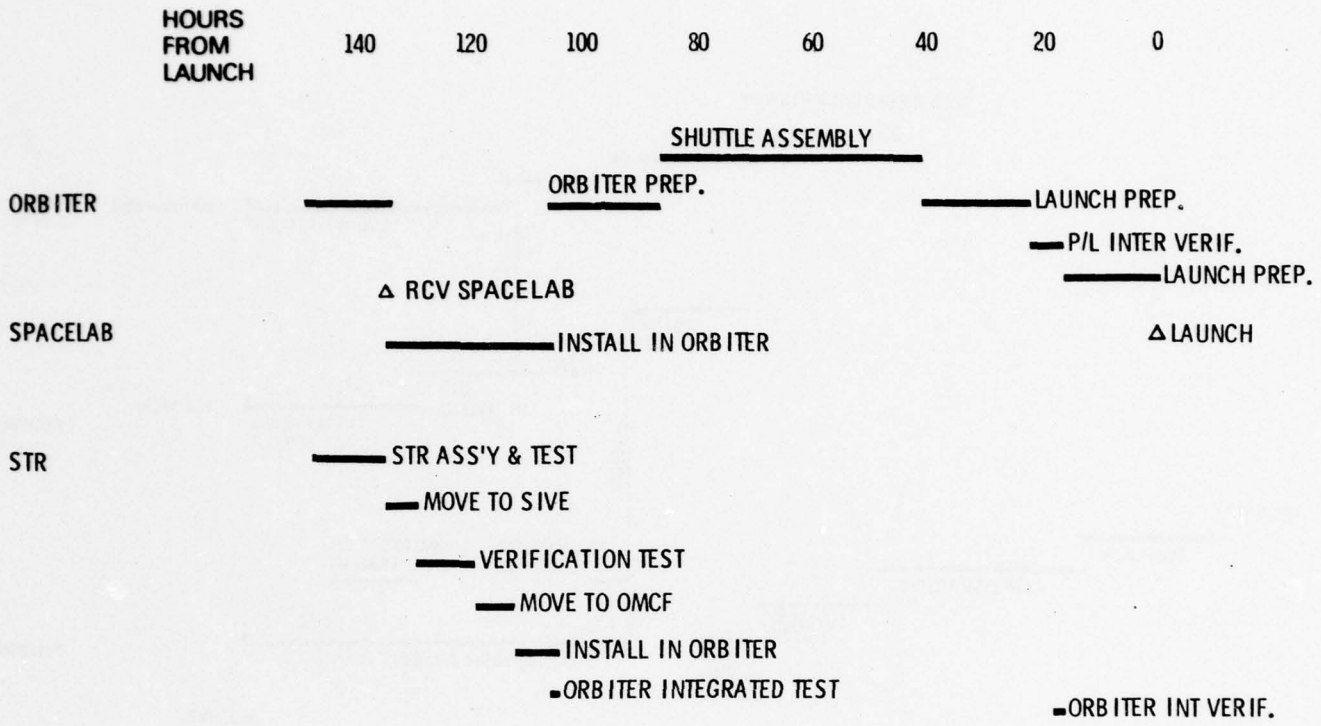


Figure 5-35. STR Timeline with Spacelab - VAFB

panel in the payload station and the copper paths from it to the payload will be verified. Also, any data required from the Orbiter will be communicated to the payload and the data transmitted from the payload to the Orbiter will be verified.

Shuttle assembly is performed in the Vertical Assembly Building (VAB) at KSC, and on the launch mount using the Mobile Service Tower at VAFB, In both cases all flight equipment (Orbiter, Solid Rocket Boosters, and External Tank) are mated vertically. Mechanical interfaces are verified sequentially throughout this Shuttle buildup sequence. Electrical interfaces are verified after completion of the final stackup.

5.4.2.3 STR Integration Timelines - Spacecraft

The timeline activities shown in Figures 5-36 and 5-37 are related to the ground processing operations of the STR with Spacecraft at KSC and VAFB, including also an IUS at KSC (the IUS is not flown on any missions launched from VAFB). The payloads at both locations are installed vertically on the launch pad using the Payload Ground Handling Mechanism (PGHM) after checkout in the PPR or PCR.

After the payload is installed, all Orbiter/payload mechanical, electrical and fluid interfaces (if required) are connected. A power-up verification of the Orbiter-to-payload interfaces is made prior to closeout of the payload bay. Final payload servicing is performed and the payload bay is inspected. These activities are followed by payload bay closure.

5.4.3 STR POST-FLIGHT OPERATIONS

Post-flight removal and ground processing operations of the STR at KSC and VAFB are shown in Figure 5-38 and 5-39. The STR operations procedures at both locations are very similar with the only differences being the name of the facility to which the STR is moved after removal from the Orbiter.

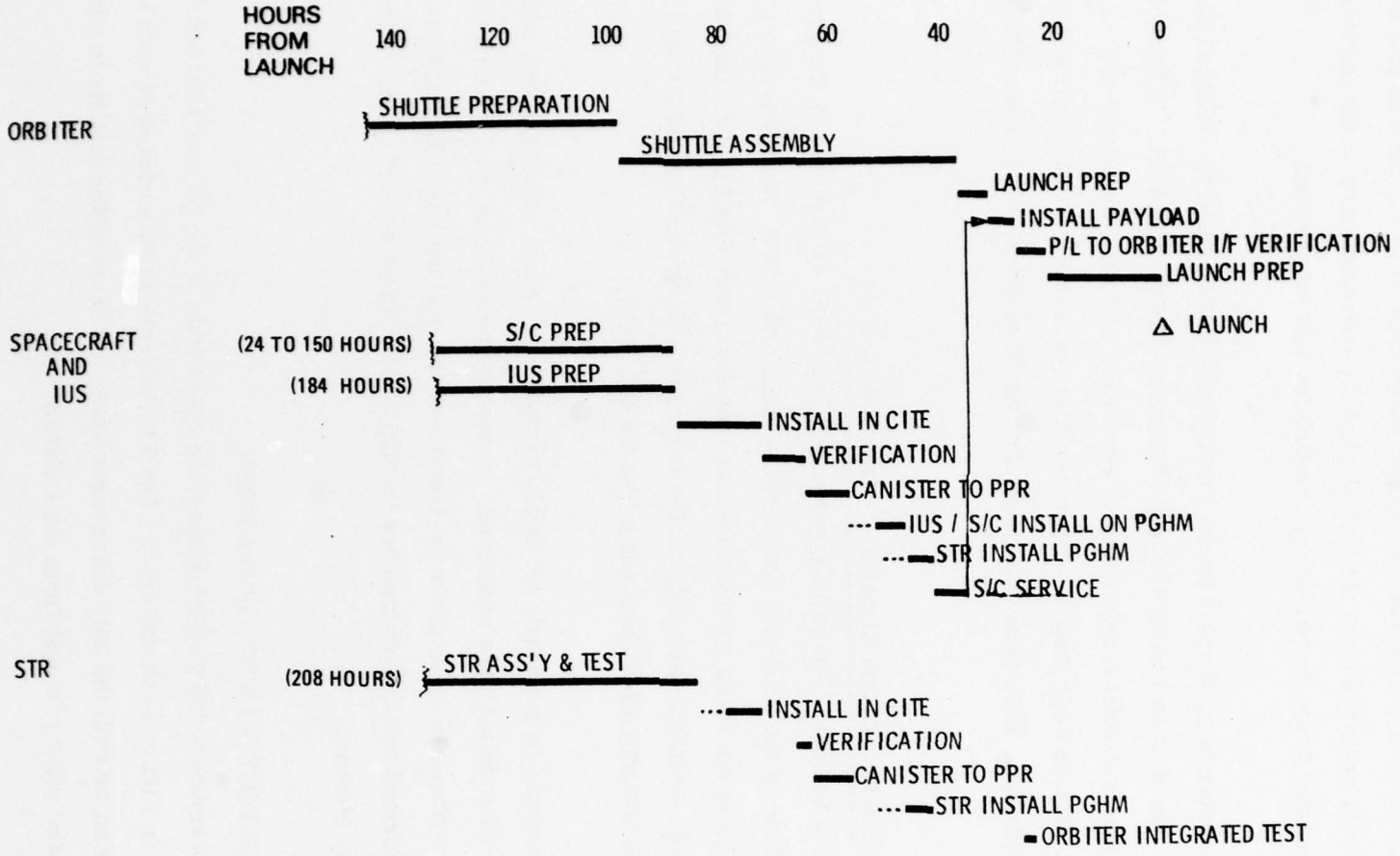


Figure 5-36. STR Timeline with Spacecraft/IUS - KSC

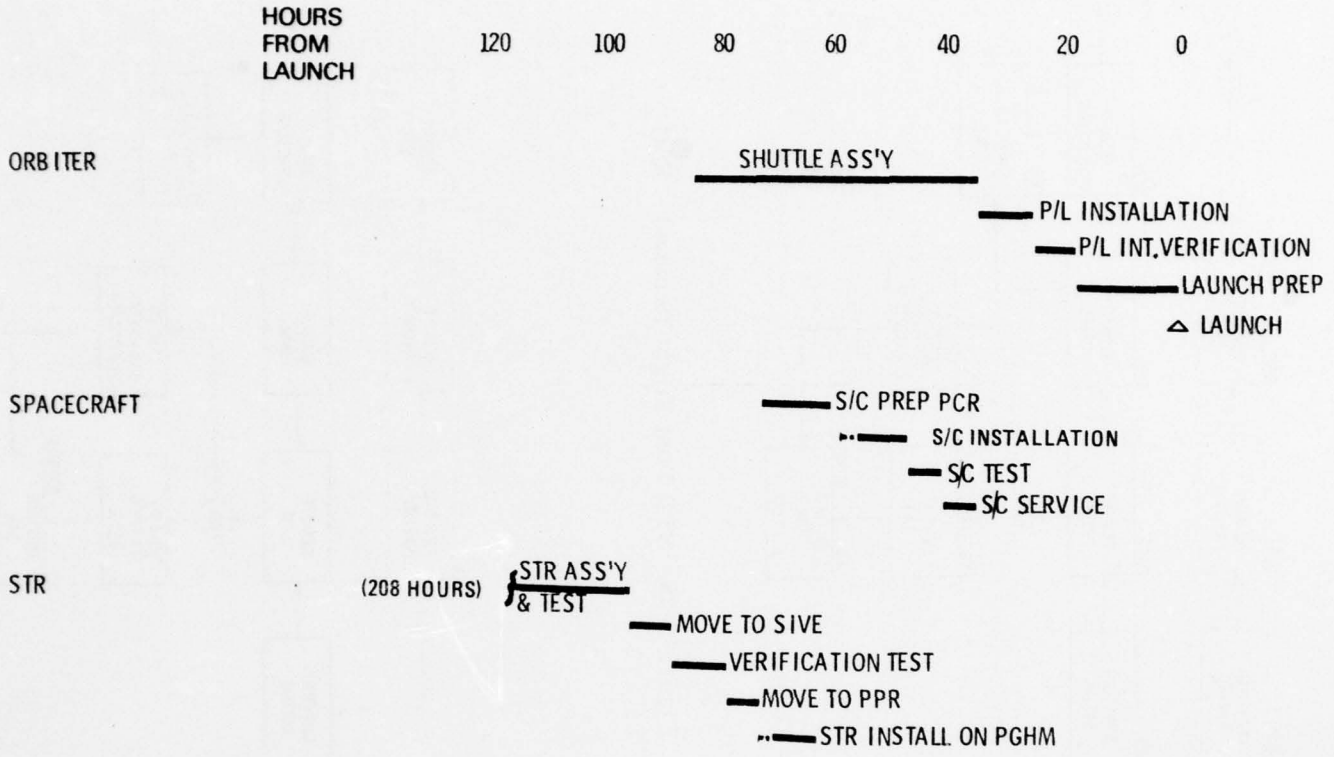


Figure 5-37. STR Timeline with Spacecraft - VAFB

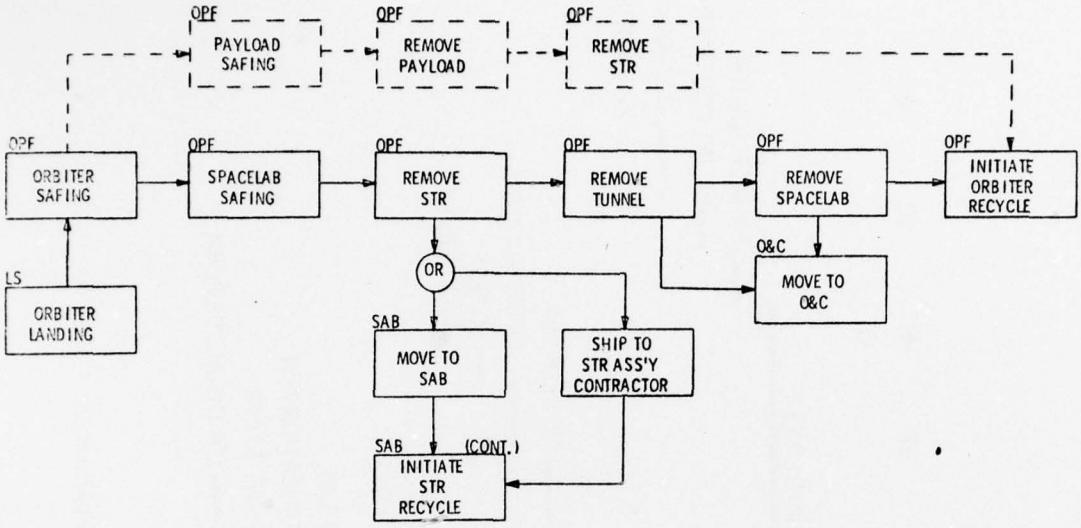


Figure 5-38. STR Post Flight Removal - KSC

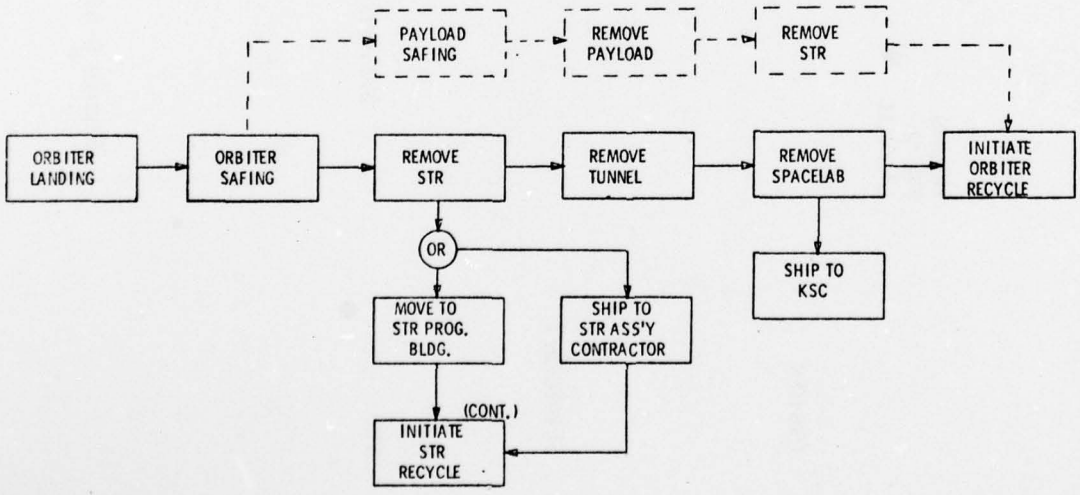


Figure 5-39. STR Post Flight Removal - VAFB

Orbiter landing operations will include the shut down of auxiliary power units within one minute after stop, connecting ground service units, crew egress, and a preliminary safety inspection for propellant leaks. Orbiter safing operations include the removal of cryogenic and hypergolic fuels, cooling and venting the interior, purging tanks and systems, and the removal of electronically stored data.

The next Orbiter operations involve payload removal preparations and final removal of the payload in order to perform various tests, inspections, and maintenance activities in preparation for the next flight. The STR is removed during this phase and transferred to the appropriate facility for refurbishment and logistic operations

The primary differences in the activity flows depicted in Figures 5-38 and 5-39 are those related to Spacelab disposition. At KSC the Spacelab is simply moved to the Operations and Checkout (O&C) building for disassembly; whereas at VAFB, the Spacelab is removed, prepared for shipment, and shipped to KSC for disassembly.

5.4.4 STR REFURBISHMENT

It is expected that the STR refurbishment activity will have an insigificant impact on the STR turnaround time cycle. The required design life of STR subsystems should minimize the possibility of their requiring any extensive refurbishment between flights. The major impact on the turnaround time cycle will be related to STR reconfiguration activities. The primary tasks to be performed as part of the flight-to-flight refurbishment activity are: inspecting, cleaning, and servicing of hardware to detect and prevent wear and the effects of aging.

Structural/Mechanical elements will be reviewed for:

1. Cracks (fatigue, stress corrosion, abnormal loads, material flaws, material property degradation)
2. Leakage (seals, joints, fasteners)
3. Debonds (peel, delamination)
4. Wear (normal, interference, threads, abnormal loading)

Refurbishment of dedicated electronic/electrical subsystems is expected to be minimal due to the use of highly reliable solid state devices and standardized hardware. Post-flight checks are made to verify their functional performance capability. Data processing system checks can be performed using diagnostic and self-check software. Tape drives and read/write heads on mass memory units, where applicable, including any high rate digital recorders, will be serviced. Functional and calibration checks will be performed on the caution and warning system interfacing hardware.

Any STR dedicated structural/mechanical elements and electronic/electrical components requiring extensive refurbishment will be removed and replaced by line replaceable units (LRU) from the spares inventory. This will minimize the impact of the refurbishment activity on the STR turnaround time cycle.

5.5 SPECIAL ACTIVITIES REQUIRED

A review of the STR/Orbiter interfaces and the integration activities indicated that no extraordinary activities are required. There are, however, some activities that are not typical of sortie or free flier missions covered in other studies conducted for DOD and NASA.

A summary of these activities is:

1. Track STS Flight Manifest, Schedule, Integration
2. Multiple Element Installation
3. LN_2 for Cryogenics
4. Storage in Controlled Area
5. Maintain Flight Ready Condition
6. Integrated installation with spacecraft/IUS or Spacelab Tunnel
7. Installation of STR Around Tunnel

5.5.1 TRACK STS FLIGHT MANIFEST, SCHEDULE, INTEGRATION

In order to take advantage of flights of opportunity, it is necessary that DOD track the STS flight manifests, schedule and missions. This is done so that when a DOD

experiment or other mission is defined with its mission and performance requirements, potential STS flights can be identified and negotiations initiated to obtain a flight commitment. A planning matrix as shown in Figure 5-40 is visualized as the type of documentation and the typical mission characteristics which affect payload acceptability for each flight. These characteristics are considered necessary to establish STR compatibility and would include such data as:

1. Launch information - site, date, time , etc.
2. Orbit parameters - altitude, inclination, attitude cycles and limitations, ephemeris, etc.
3. Mass Properties - unused weight available for additional payload, e.g., location of total payload.
4. Volume - open space which represents potential mount points for STR, the trunnion points available on keel and longeron
5. Orbiter services available - power, electrical energy, telemetry, data recording, C&W and command
6. Aft flight deck console room
7. Crew time
8. RMS availability -as a function of time
9. Cargo list - Spacelab and experiments, Spacecraft, Upper Stages, LDEF, etc.
10. Special limitations.

The matrix is constantly updated so that available space is identifiable for as far into the flight program as payloads are identified. This data is applicable to the STR as a "piggyback" or secondary payload to obtain maximum cost effectiveness from the STS. If an experiment has sufficient priority, it can be flown as a primary payload with the ground rules that apply to them.

Detail schedules for integration thru launch are also kept for each flight as those schedules become available. They will be used to determine hardware need dates and Orbiter and payload integration schedules for the STR's.

STS FLIGHT / FLIGHT DATA	7	8	9	10
1. LAUNCH INFORMATION				
1.1 SITE				
1.2 DATE				
1.3 TIME				
2. ORBIT PARAMETERS				
2.1 ALTITUDE				
2.2 INCLINATION				
2.3 ATTITUDE CYCLE				
2.4 EPHEMERIS				
3. MASS PROPERTIES				
3.1 WEIGHT (UNUSED)				
3.2 C.G. OF TOTAL CARGO				

Figure 5-40. STS Flight Payload Data Matrix

5.5.2 MULTIPLE ELEMENT INSTALLATION

For very large payloads which will be mounted on the STR, it is possible that two STR structures separated by some distance will be used. In this instance, three elements will be mounted to comprise a total STR payload. These elements would be:

1. A STR complete with subsystems
2. A second STR, structure only
3. A large payload, i.e., which must be supported at two points in a stowed position for launch and landing. A long telescope would be an example. (See Figure 5-41).

An Example:

For this case, special methods of attaching the STR payload to the two STR structures is required.

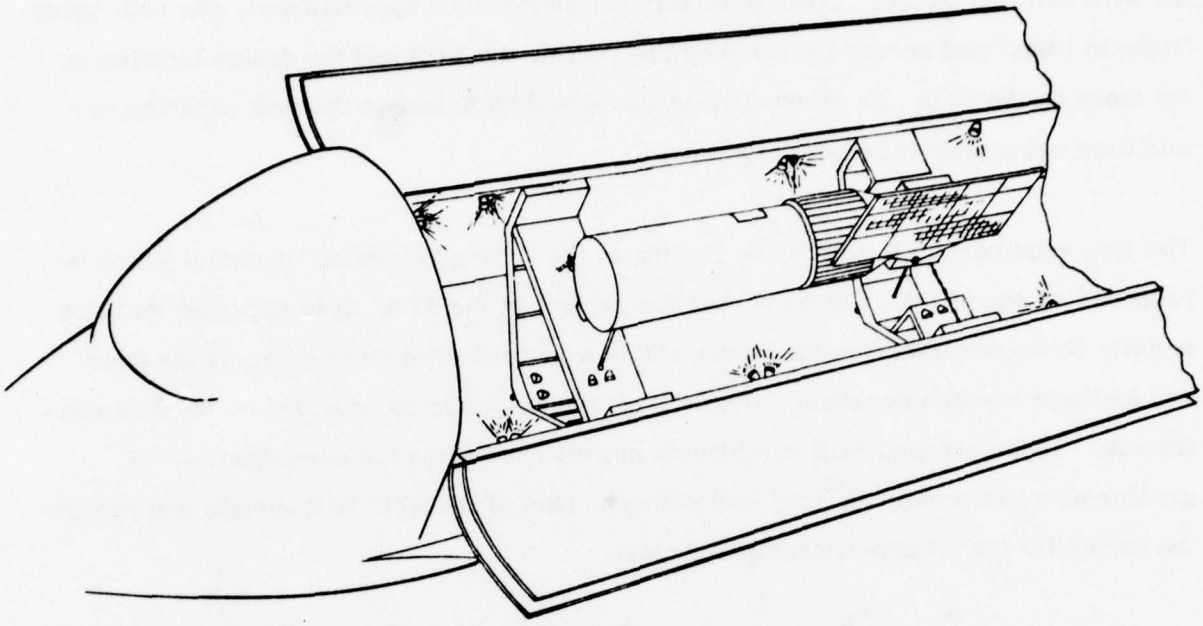


Figure 5-41. Multiple Element Installation

5.5.3 LN_2 AND CYROGENICS

Experiments 2, 4 and 9 require cryogens to cool detectors in their sensors. The cryogens are as follows:

1. Solid neon and solid methane
2. Liquid helium

The solid neon and solid methane are loaded as gases and solidified in place using liquid helium. The total cryogen charging and cooling sequence is done using ground equipment provided by the experiment contractor. The full cryogen charge is estimated to last without loss for up to ten days in the ambient ground launch environment and without necessity for renewal as long as the experiment is not operated. As a backup, provision must be made to accomplish final cryogen charging prior to launch to ensure a fully charged condition at launch.

The liquid helium is loaded into the tanks from an external source provided as AGE equipment by the payload contractor. The liquid helium will totally vaporize in

about ten days at ambient temperature. The liquid helium must be loaded on the pad in the PCR prior to launch. Loading time is estimated to be approximately one hour using "drag in lines" and access provided by the PGHM, the PCR and the design location at the tanks on the STR. An alternative to this would be to double the tank capacity, but this does not appear to be cost effective.

The LN_2 requirement is to provide cooling to the "change of phase" material which is proposed as one of the methods of thermal control at the STR. It is expected that this activity be accomplished on the launch pad in a manner similar to charging the Neon and Methane cryogen systems. The AGE equipment would be provided by the STR contractor. These cryogen AGE equipments are all candidates for commonality. If greater usage is predicted, they could become part of the STR AGE system and provide the capability for all experiment contractors.

5.5.4 STORAGE IN A CONTROLLED AREA AND MAINTAIN FLIGHT READY CONDITION

The STR/payload can be assembled and tested without having been assigned to an STS flight. The STR/payload assembly in this case will be stored to await a "flight of opportunity". Such storage could be in an STR shipping container if it were initially designed to house a variety of large experiments mounted on the STR.

It is more likely that a storage room will be used so that large STR experiments and more than one STR assembly can be accommodated. The testing to determine flight readiness status and any refurbishment or updating can be accomplished within the storage room. However, the storage room must provide the proper environment control and contamination levels. This room will be part of the facilities space required for the on-site STR integration activity.

5.5.5 INTEGRATED INSTALLATION

A possible mode of installation occurs with mounting of a strongback/bridge configuration of the STR around the spacelab tunnel or installation of a strongback/shelf configuration of the STR around a spacecraft cantilevered off a support cradle or an super stage

(IUS or SSUS). In these cases, the strongback is separated from the bridge or shelves and is installed in the Orbiter bay. The tunnel or spacecraft is then installed within the strongback configuration. The STR installation is then completed by installing the bridges or shelves and attaching them to the strongback.

5.5.6 INSTALLATION OF STR OVERSPACELAB TUNNEL

In some instances, it might be desirable to install the STR into the Orbiter around the Spacelab tunnel after the tunnel has been installed. To accommodate this the strongback and bridge will be installed separately.

The strongback is installed first using the loading fixture designed to rotate and translate its load. This activity is shown in Figure 4.5-2. This operation can be done for horizontal or vertical Orbiter loading positions. The fixture is designed to operate with the Orbiter Horizontal Loading Fixture (in the OPF at ETR and OMCF at WTR) and with the Payload Ground Handling Mechanism (PGHM) in the PCR (at ETR and WTR). After the strongback has been installed and latched to the Orbiter pyaload bay longerons at the trunion points, the bridge structure is mated and secured to adjustable support fittings on the strongback. Mating of STR internal and STR to Orbiter electrical connectors completes the installation of the STR.

5.6 GROUND EQUIPMENT AND FACILITIES

A summary of the mechanical and electrical AGE is given in Table 5-14 and 5-15. This equipment is utilized in the assembly and testing of the STR and experiments and in the preparation for and installation into the Orbiter bay.

An automated checkout console is utilized to minimize the test time which in turn reduces the turnaround times. The ground station equipment is used to keep the testing independent of the prime mission ground station. It is possible to climinate this equipment and utilize the mission ground station with hardware or airlink as required.

In summary, no extraordinary equipment is required to support the STR.

Table 5-14. Mechanical Equipment List

- STR System Test Dolly
- STR Transporter
- STR Sling Set
- STR Loading Fixture
- STR Vertical Ass'y Fixture (Tunnel)
- Strongback Support Stand (LH & RH)
- Bridge Support Stand
- STR Shipping Canister
- Universal Sling Set
- Panel Handling Fixtures (6)
- Alignment Fixture
- Alignment Targets
- Sensor Stimulation Supports
- Alignment Target Supports
- Gas Service Unit (GN_2)
- Thermal Stimulation Test Set (Heat Pipes)
- Accountability Kits
- Turn-over Fixture
- LN_2 Service Unit
- Assembly Tool Set

Table 5-15. Electrical Equipment List

- Automated Checkout Console
  - System Test Set
  - Electrical Power System Test Set
  - 28 VDC Power Supply
- ACS Stimulator
- Break out Boxes
- RF Power & Frequency Measurement Equipment
- Battery Servicing Equipment
- Ground Station Equipment Simulator
  - RF Receiver
  - Brush or Line Recorders
  - S-Band Ranging Equipment
  - Analog Tape Recorders
  - Command Console
  - Ground Station Computer
  - Computer Auxiliary and IO Units
  - Wideband Receiver

A review of the twelve baseline experiments indicates that no special facilities are required at KSC or VAFB, except the space for STR/experiment assembly and test. A summary of the space required is as shown below in Table 5-16.

This total area is determined assuming one STR system in assembly or test, two in storage and one awaiting assembly and test. A total of four STR's can be on the premises at one time with the accompanying payloads.

A list of required facilities is given in Table 5-17.

Table 5-16. Facilities Space for STR

Usage	Area in Feet^2
STR/Experiment Assembly and Test	1200
Alignment of Sensors & Experiments	600
Experiment Receive, Inspect, Checkout	450
STR/Experiment Storage	400
Mechanical GSE Storage	600
Electrical GSE	400
General Storage (Spares, etc.)	200
Offices	900
Total	4750

Table 5-17. Facilities List

- STR Assembly and Test Area
- STR Storage Room
- Spares Storage Room
- Alignment Area
- LN_2 Supply
- Experiment Requirements as Necessary

5.7 INTEGRATION SUMMARY

Table 5-18 summarizes the results of the integration and operations study.

Table 5-18. Integration Summary

• Orbiter Timelines Minimally Extended
• STR Similar to Automated Payload
• No Extraordinary Equipment or Facilities Required
• Dedicated STR Processing Area Recommended

Most of the integration activities can be conducted off-line with respect to the Orbiter and its payload. The off-line activities are the same as for other typical payloads for sortie or free flier missions. They include Orbiter interface verification testing, installation and attachment, Orbiter/STR compatibility checks and an STR/experiment functional checkout to verify flight readiness. If these STR activities were done in series with similar activities for the other payloads, the timeline extensions would be approximately eleven hours. If they are all done in parallel, the timeline extension is zero hours.

The facilities provided at the launch sites satisfy the requirements of the STR and no special facilities are required. It is necessary to provide space for a dedicated STR assembly and test area. It is expected that this area will be in continuous use throughout the first shuttle decade.

5.8 KEY ISSUES

The key issues pertain to areas which must be negotiated with NASA. These issues affect the quick reaction time due to their requiring an interface with the Orbiter, the extent of testing for flight certification, or the time required to plan and accomplish the STR integration activities. The key issues are listed in Table 5-19.

Table 5-19. Key Issues

• RF in Orbiter Bay
• Crew and MCC Override (Depth)
• STR Integrated Verification Testing
• Last Possible Date for Flight Commitment
• On-Site or Off-Site Integration of STR

In early Shuttle integration studies, any RF in the Orbiter bay was deemed to be unacceptable. Later studies showed a softening of this limitation but it is still not clear. The deployment/use of STR mounted antennas for integration with the SGLS must be investigated and acceptance of their use obtained.

From the safety standpoint, the crew should have some control over the Orbiter payloads. This control could be, at the least, a turn-on and turn-off capability. Other potential requirements include such items as jettison of the total STR assembly or parts of the STR assembly. The extent and depth of crew control will have to be examined for each experiment flown. The participation of the STS Mission Control Center in override would be included in consideration of the crew control of STR payloads.

It is assumed in this study that the Orbiter interface verification testing is done with the other Orbiter payloads. In cases where there is obviously no possible effect of the STR on other payloads, it might be convenient to verify the Orbiter/STR interfaces separately, particularly if the mobile IVE concept is adopted. In other cases, such as re-flight of an STR/payload, if the Orbiter/STR interface verification tests were not required, operations costs would be reduced. The extent of STR integrated verification tests must be determined and verified.

The quick reaction time for the flight of opportunity mode is a basic requirement of the STR concept. Although physical integration can be accomplished within the

current Orbiter payload integration timelines, the integration planning and verification of flight readiness can take one to six months, depending on the STR complexity and the other payloads involved. It is necessary to determine if there are STS factors which limit the time before launch for commitment of an STR flight.

The on-site integration and test of the STR and experiments is recommended to minimize STR logistics costs and shorten the quick reaction time for last minute STR flights. This requires a permanent dedicated facility area at KSC and VAFB. A trade-off should be made considering such things as available space in existing buildings, need for a new building, the off-site location and cost of off-site integration by an integration contractor. This trade-off can be used to determine the location and number of integration facilities.

SECTION 6

STR MISSION MANAGER CONCEPT

The STR role is to enable DOD to fly experiments, prime high risk equipment and prime mission equipment on Shuttle. The STR is considered as a payload sharing the Shuttle with primary cargo of NASA or DOD sponsorship. The primary cargo can be spacelab, sortie mission or free flier missions with or without upper stages. This section deals with management of the integration of the STR and its payloads into the STS system.

6.1 STR MISSION MANAGER

The usage and operation of the STR will be managed by a dedicated integration team directed by a Mission Manager. He will direct the STR activities, monitor the performance of those activities and make necessary decisions to insure successful completion of those activities.

The task of the Mission Manager is to integrate payloads into the STR and to integrate the STR payloads assembly into the STS. Since the STS integration should be started as early as possible in any payload program, assigning one individual, the STR Mission Manager, to payload and STS integration assures the best chance of procuring and meeting STR flight dates on the STS.

The STR Mission Manager approach stresses centralized authority, streamlined procedures, and minimum documentation. Each of these major elements reinforces the others to create a system which stresses individual responsibility and team work as opposed to dispersed responsibility and paperwork. Like any such system, its success is a function of the people involved, but given a reasonable level of motivation and competence it offers the promise of low cost and efficient operations.

Individually assigned and visible responsibilities - with matching authority - is the keystone of the STR management approach. The Mission Manager is established as the

single point of authority for the STR to (1) direct STR personnel and to (2) make interface agreements with payload contractors and sponsors (SPO's) and STS Mission Managers. Since a similar Management Concept is being considered for STS, the STR Mission Manager approach should correlate well into the overall STS flight management system.

The STS Mission Manager is the single central authority on cargo for STS flights. He derives his authority from the Prime Payload Sponsor (usually one of the NASA centers or DOD SPO's) who "buys" the Shuttle flight. The role of STS and its support teams is to take direction from the STS Mission Manager and support him to the capabilities and the constraints (physical, functional and safety) of the STS program. The Prime Sponsor "sells" space to other users through the STS Mission Manager. He integrates user requirements and forms contract agreements between the Prime Sponsor and the "other users."

The role of the STR Mission Manager is that of "other user." The STR Mission Managers concept is shown in Figure 6-1. Also shown in the relationship of STR Mission Managers to STS Mission Managers. The responsibilities and relationships between the STR and STS teams is shown in Figure 6-2. The accomplishment of many combined activities in the payload integration scenario will require control by the STR Mission Manager of experiment personnel and control by the STS Mission of STR personnel. No attempt was made to describe in detail just how this type of control is to be implemented, since that was beyond the scope of the present study. However, as shown by dotted line, in Figure 6-2, some sort of control is necessary for efficient operations. Although this study does not cover this area in detail, this relationship is critical if low-cost operations are to be accomplished.

6.2 STR MISSION ACTIVITIES SCENARIO

The STR System is used to fly experiments and mission equipment for DOD payload sponsors. The STR Mission Manager operates the STR System and manages its usage. The STR System usage must be as flexible as the STR hardware. The usage of STR can be initiated for any payload at any time in the payload development cycle or even

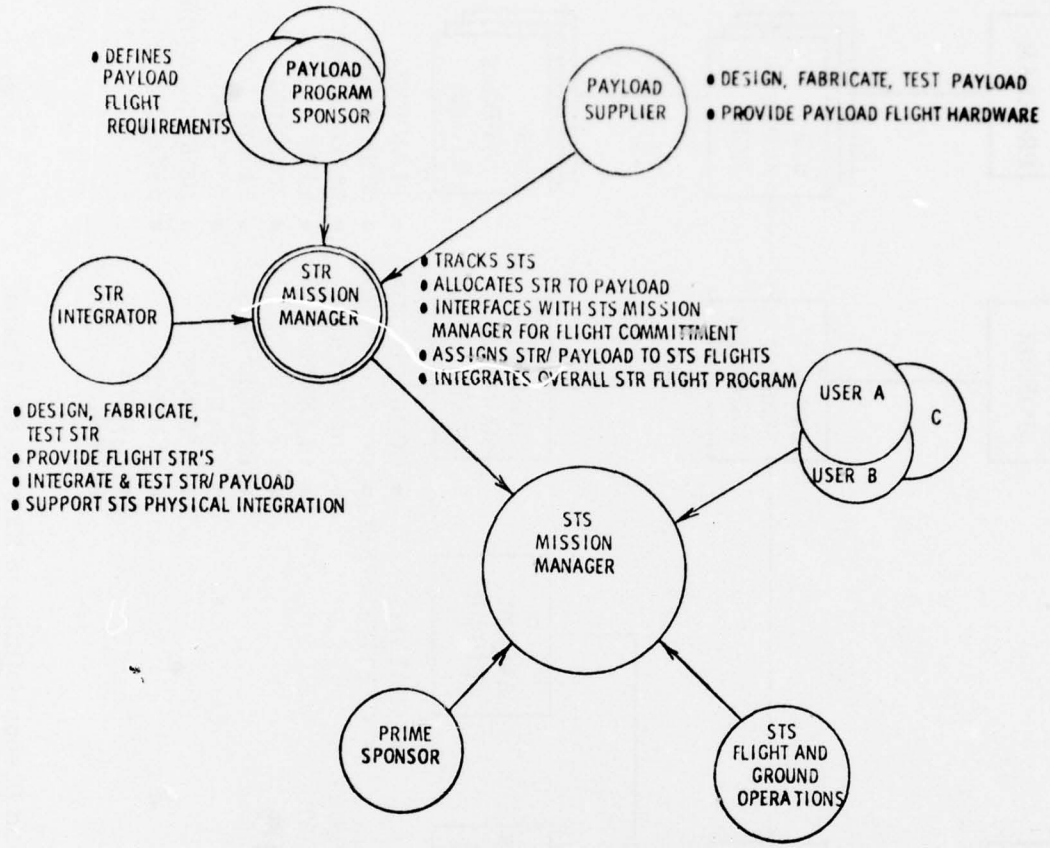


Figure 6-1. STR Mission Manager Concept

after completion of that cycle. The STR Mission Managers activities for a specific payload begin with a request from a DOD payload sponsor for flight. The activities cover the following mission phases:

1. Payload integration into STR
2. STR integration into STS

These activities are further defined in Figure 6-3. As shown in the figure by horizontal arrows, the timing of integration activities with respect to the other activities can vary as required by program priorities, timing etc. The activities joined by vertical arrows must occur serially.

The following paragraphs discuss the details of the STR payload management scenario shown in Figure 6-3 for Shuttle integration activities. The seven phases are discussed

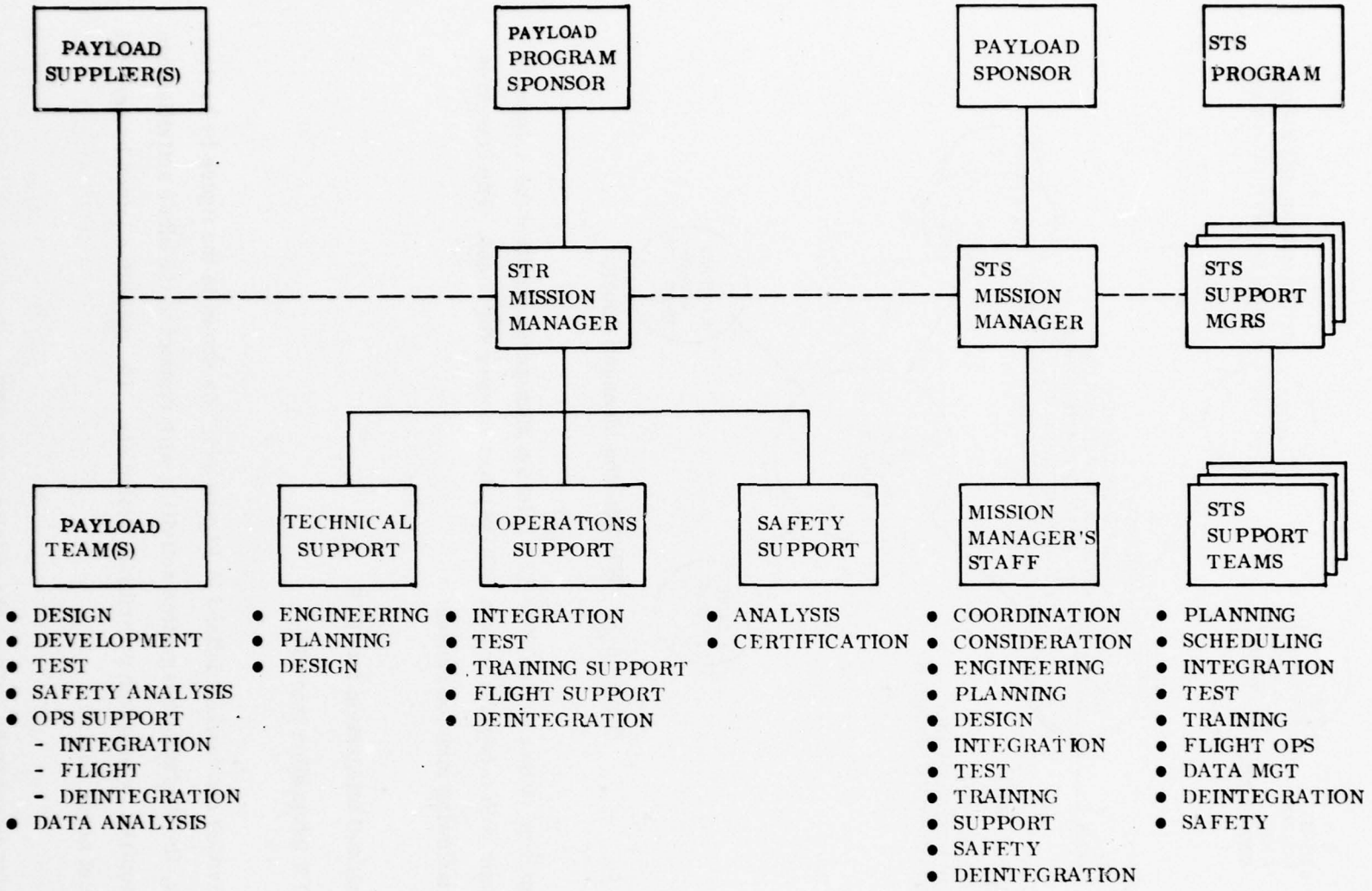


Figure 6-2. Relationships and Responsibilities of STR and STS Teams

separately and later sections of this report present descriptions of the interface documentation and Mission Manager staffing requirements derived from the scenario. Much of these activities are described in detail in Section 5. The Mission Approval phase includes the request for flight from a DOD payload sponsor and the activities taken by the STR Mission Manager to get space allocated by the STS Mission Manager on a specific STS flight. This request can come early in the payload development phase or later, even after the payload development is complete. The STR Mission Manager by proposal and /or STS Form 100 requests STS flight space. He can request and have allocated to him space and accommodation resources on a specific or generic basis at any time; i.e., a flight reservation for an STR with a still to be developed payload may be obtained. This latter approach can provide for "last minute" flight requests.

The second phase is the integration of the payload into the STR. These activities include:

1. Establishing payload/STR interfaces and requirements
2. Configuring STR for the payload
3. Top level installation and assembly drawings
4. Detail design and fabrication of any required equipment such as insulation blankets
5. Analyses updates for the STR with the specific payload
6. Software update for the specific payload
7. STR/payload interface verification using STR simulator and mounting hardware

Much of this activity, primarily the analyses interface verification and software, are applicable to the STR/STS integration activities. The STR/STS integration activities can occur serially or in parallel with the payload/STR integration as desired for convenience, low cost or as limited by data availability or program schedule. In general, the earlier this activity is initiated, the easier it is to accomplish the end

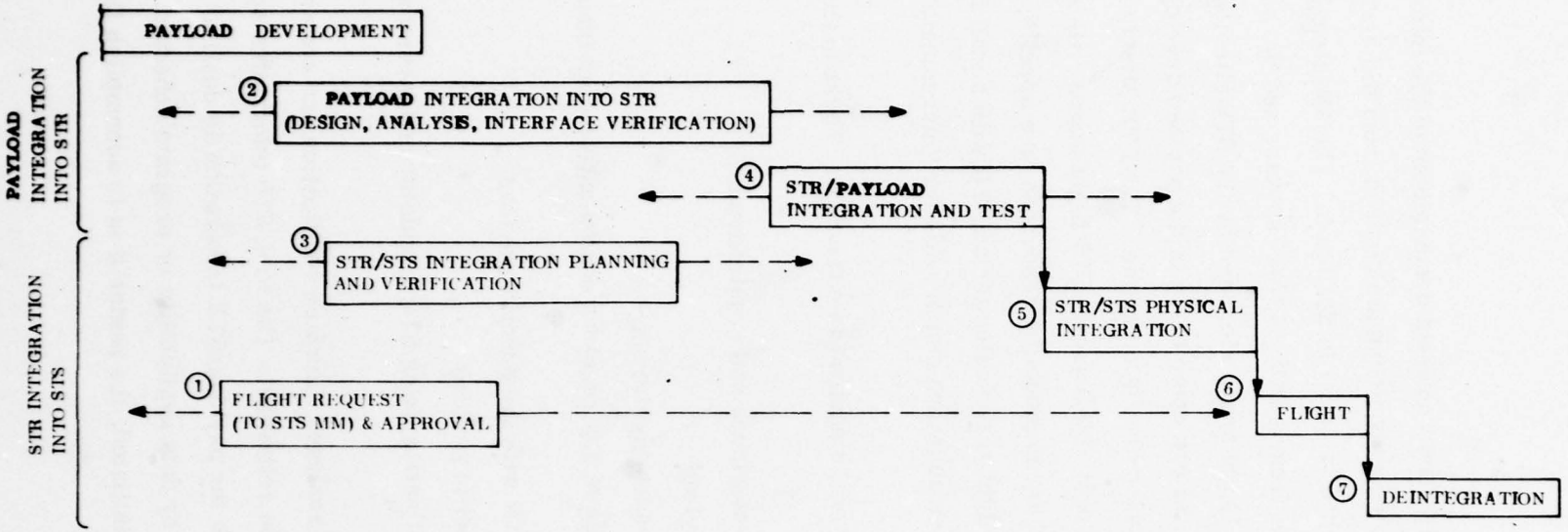


Figure 6-3. STR Mission Activities Cycle

goal of acceptability of an STR/payload assembly for flight. In this phase compatibility of the STR/payload with the STS and other payloads must be verified by analysis, test, etc.

The fourth phase is the physical integration and test of the experiment and STR. This activity is carried out at a dedicated STR preparation area at the ETR or WTR launch site. It may include a post-test storage period while awaiting a flight or flight approval. The fifth phase, physical integration of the STR/experiment assembly into the STS, includes the following activities.

1. Pre-installation testing (CITE or SIVE to verify STR/STS and other payload interfaces)
2. Final servicing
3. Installation into STS
4. Post-installation STR/STS interface verification

The flight phase requires the support necessary to operate the STR and the experiment during launch to carry out the experiment mission and support the STS mission and safety of flight activities.

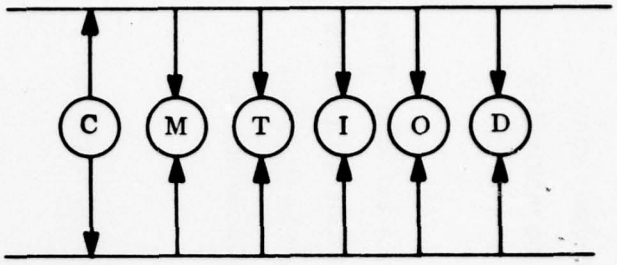
The last phase is deintegration and includes the safing, inspection and removal of the STR from the STS and returning it to the dedicated STR preparation area. At that point, the payload is removed or prepared for a reflight. In the former case the STR is prepared for recycle with another payload.

During all of these phases, the STR teams are under control of the STR Mission Manager. In accomplishing his task the Mission Manager also interfaces with other teams involved in an STS flight.

Figure 6-4 and 6-5 illustrate one of the methods used to define and summarize the number and type of interactions taking place in the scenario. An important feature

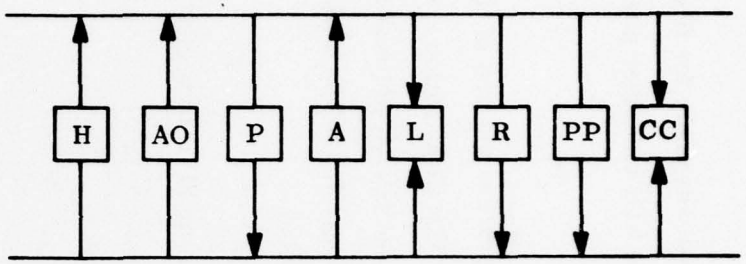
COMBINED ACTIVITIES

C	CONSULTATION & COORDINATION
M	FORMAL MEETINGS
T	QUALIFICATION & TRAINING
I	INTEGRATION, TEST, CHECKOUT, & SERVICING
O	FLIGHT OPERATIONS & SUPPORT
D	DISASSEMBLY, DEINTEGRATION, & STRIPOUT



DOCUMENTATION

H	HANDBOOKS, BROCHURES, & UPDATES
AO	IFPs & AFOs
P	PROPOSALS
A	APPROVALS & ACCEPTANCES
L	LEGAL CONTRACTS
R	REQUIREMENTS
PP	PLANS & PROCEDURES
CC	CONTRACT CLOSEOUTS



DELIVERIES

E	EQUIPMENT
D	DATA

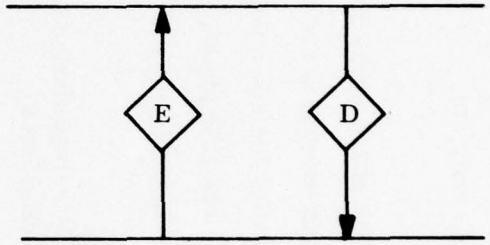
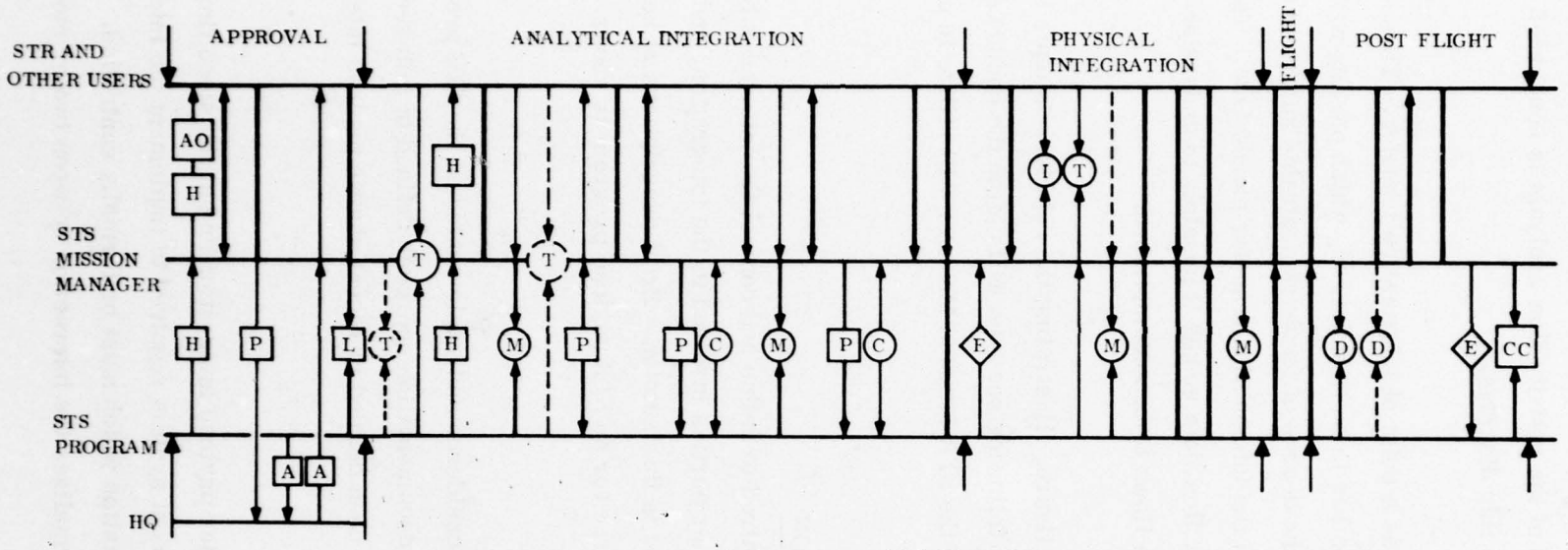


Figure 6-4. Types of Interactions



KEY

- PRIMARY INTERFACE
- MAJOR EVENTS
- OPTIONAL PARTICIPATION
- COMBINED ACTIVITIES
- DOCUMENTATION
- DELIVERIES
- STR ACTIVITIES

PARTIES INVOLVED	ESTIMATED NUMBER OF STR-STS INTERACTIONS
STR - MISSION MGR	54 - 59
STR - STS	6 - 1
TOTAL	60

Figure 6-5. Summary of Interactions

of this type of analysis was the indication of relationship between the parties, and the direction of flow of information. Much of the activity flow analysis is concerned with the content of the interactions shown on this diagram.

Each interaction identified was treated as a point of information transfer. Thus an evaluation was made of what information has to be transferred, which parties were involved, the direction of the flow and the degree of feedback desirable and/or necessary, and the appropriate form which the information transfer should take. This type of analytic treatment of the activity flow data makes it possible to understand the systematic processes necessary for the flows to be accomplished.

STR interactions are highlighted on the figure, illustrating the relatively simple interfacing for an autonomous payload. The following sections will define the documentation and organization requirements that the STR Mission Manager will utilize to carry out his tasks.

6.3 STR INTERFACE DOCUMENTATION

The major interface documentation required to define and control the interfaces between the STR Mission Manager and other parties involved in the integration, and operation of a shuttle cargo is discussed in this section. Brief descriptions of key documents are given, and a document tree for an STR payload program is shown.

6.3.1 DOCUMENTATION APPROACH

The general approach currently being considered and implemented on the STS programs is the basis for definition of the documentation approach defined in this section for STR programs. This approach has been defined in recent shuttle payload integration studies.

One of the ways to reduce costs of orbiter payload integration and at the same time reducing the complexity of the management system required to implement the integration is to reduce the interface documentation which must be formally controlled. The introduction of documentation which formalizes the interaction between two or more

parties establishes requirements for data which must be developed and passed between the parties. Production of data can become an end product without careful assessment of the costs of production. In a number of major programs large shares of total program costs are set aside for data and the resources needed to produce it. To prevent this, management data is reduced to the level which is minimally required to conduct and control the inter-party interactions.

A number of techniques are employed to accomplish this goal. One of the first is to establish two-party relationship where significant agreement must be reached. The results in a reduction of the number of parties which must be bound by the accompanying documentation, and therefore reduces its complexity and the costs of its production and maintenance. An important result of applying this technique is the accompanying reduction in the number of personnel required to review and approve the documentation and to produce supportive analyses. Application of this technique is shown in Figure 6-6.

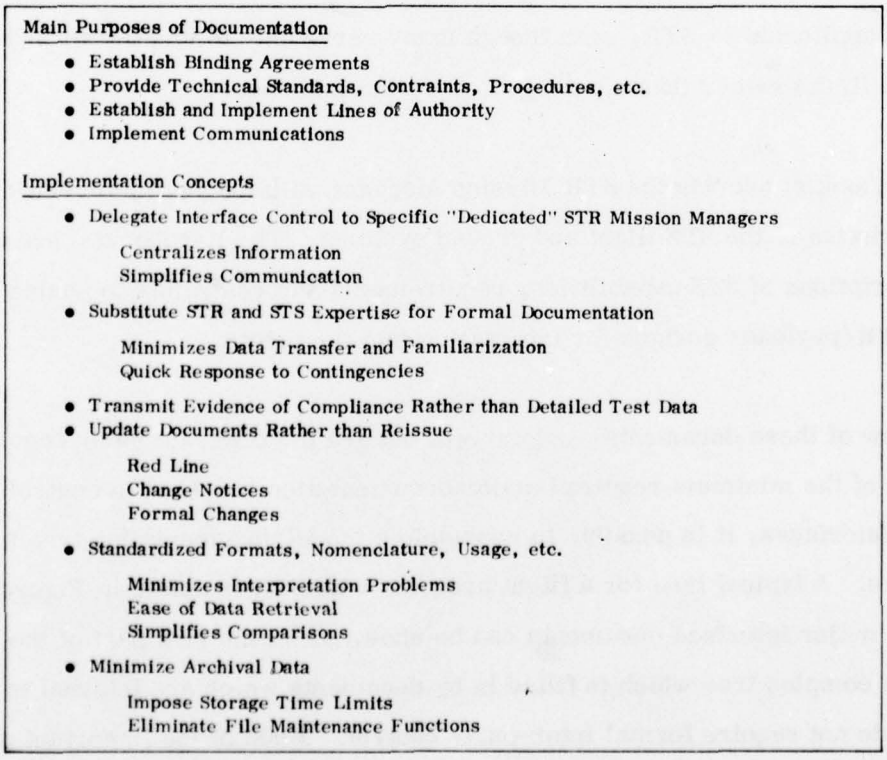


Figure 6-6. Suggested Approach to Documentation Management

Other methods utilized to reduce inter-party, formal documentation and its costs include the use of standardized formats; the use of updated or combined documents rather than new documents; the minimization of archival data; reducing the number of parties which must be involved in decision-making; the use of compliance checklists rather than full analytic reports; and reliance on personnel responsibility as a substitute for formal control documentation.

The STS users can refer to a number of STS documents which:

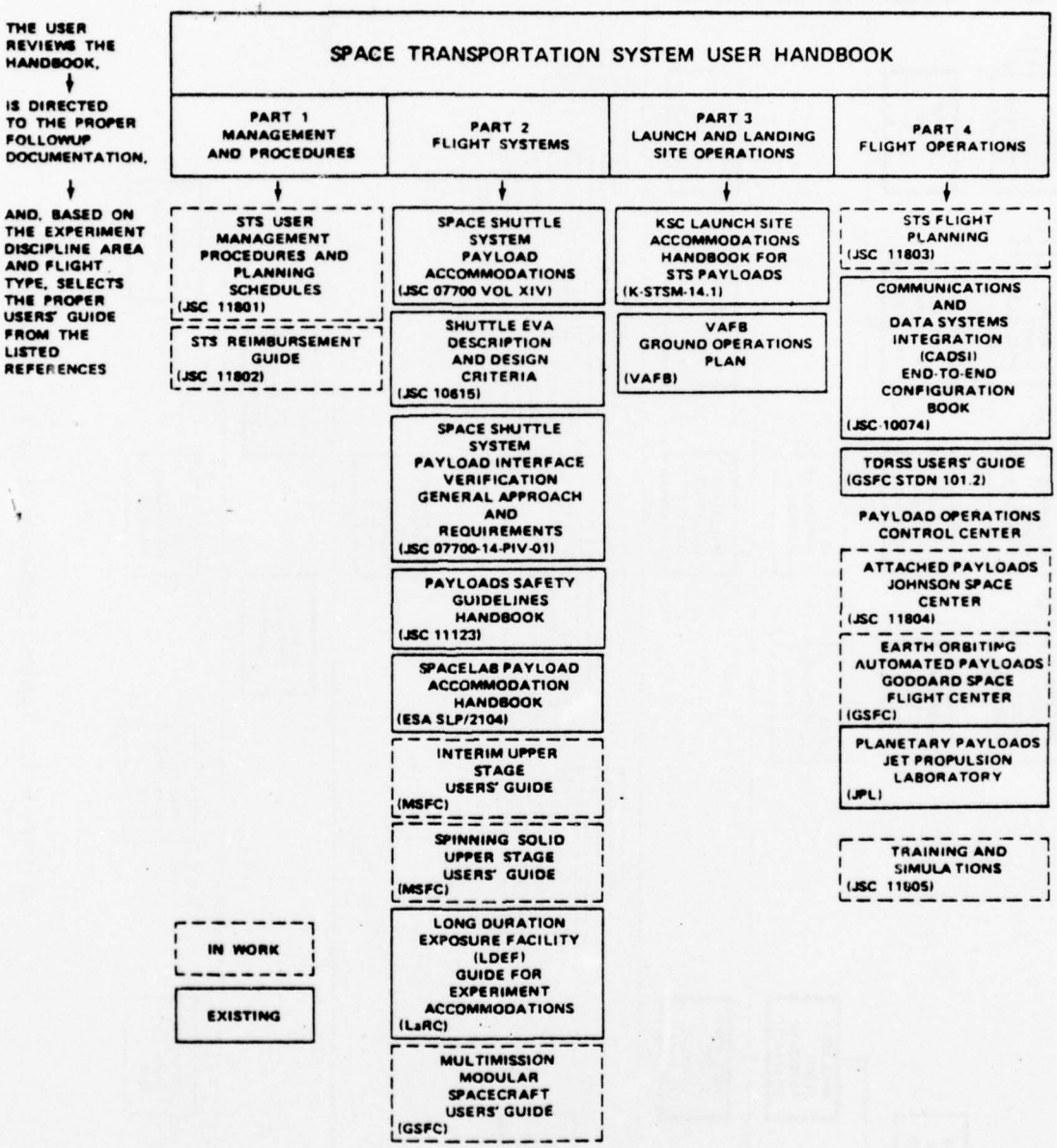
1. Provide guidance for design, integration, launch and flight
2. Define STS services available
3. Specify STS limitations
4. List compliancy requirements
5. Give guidelines for program planning and flight costs

A summary of the major STS documentation is given in Table 6-1. Each of these documents is applicable to STR, even though many serve only to identify fit and function interface limits rather than a potential operating interface.

The STS Handbook(s) provide the STR Mission Manager with in-depth information on the characteristics of the STS flight and ground systems. The Handbook(s) present detailed descriptions of STS capabilities, requirements and restraints to enable him to prepare STR/payloads designs for integration into the orbiter.

From a review of these documents and current integration program study reports and the definition of the minimum required major documentation required to control organizational interfaces, it is possible to assemble a model documentation tree for a given program. A typical tree for a flight program with STR is shown in Figure 6-7. The required major interface documents can be shown as an integral part of the larger, more complex tree which is filled in by documents which are internal to the parties, and do not require formal inter-party control. Much of the important data

Table 6-1. STS Documentation Summary



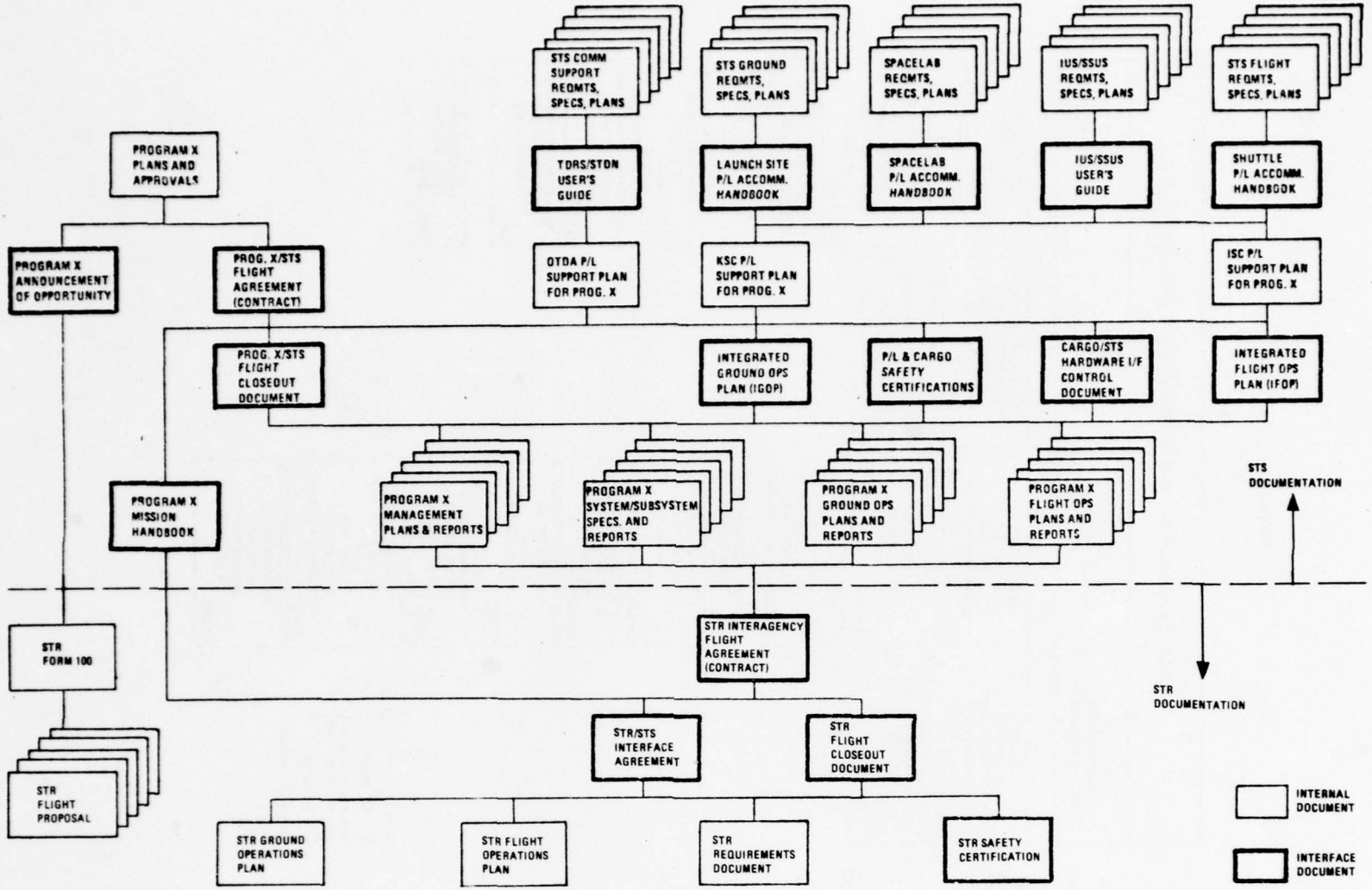


Figure 6-7. Typical Documentation Tree Program X

required to implement a given payload program can be in this state, where responsibility for its continued accuracy is in the hands only of the party which generates the data. This reliance on personal responsibility is a key to establishing low-cost payload program management.

From the foregoing document reviews and the STR/STS interface documentation definition, two categories of documents are identified:

- Controlled interface documents which are developed by the STR Mission Manager or the STS Mission Manager, are signed off by both and continually updated as required.

- Uncontrolled documents generated by the STR Mission Manager and used to transfer data to the STS Mission Manager for use in overall cargo integration and safety analysis and documentation. There will also be correlative documentation generated by the STS Mission Manager and provided to the STR Mission Manager to accommodate the needs of STR programs. These documents will augment the generic guidance and compliance documents shown in Table 6-2 and will be specific to the particular STS flight mission involved. Some of the documents shown in the upper portion of the documentation tree in Figure 6-7 fall into this category.

6.3.2 KEY STR DOCUMENTATION

A list of the nine major documents identified which must be generated for each STR/experiment flight is shown in Table 6-3. Of these nine, six are generated by the STR Mission Manager. Four of the documents are controlled interface documents. These are:

Controlled Interface Documents

Document	Responsible Party
• STR Interagency Flight Agreement	STS Mission Manager
• STR Interface Agreement	STS Mission Manager
• STR Safety Certification	STR Mission Manager
• STR Flight Closeout Agreement	STS Mission Manager

The relationship of interfaces between STR Mission Manager, the STS Mission Manager and the STS Flight and Ground Operations Teams is shown in Figure 6-8.

Table 6-2. STS Mission Manager Documents

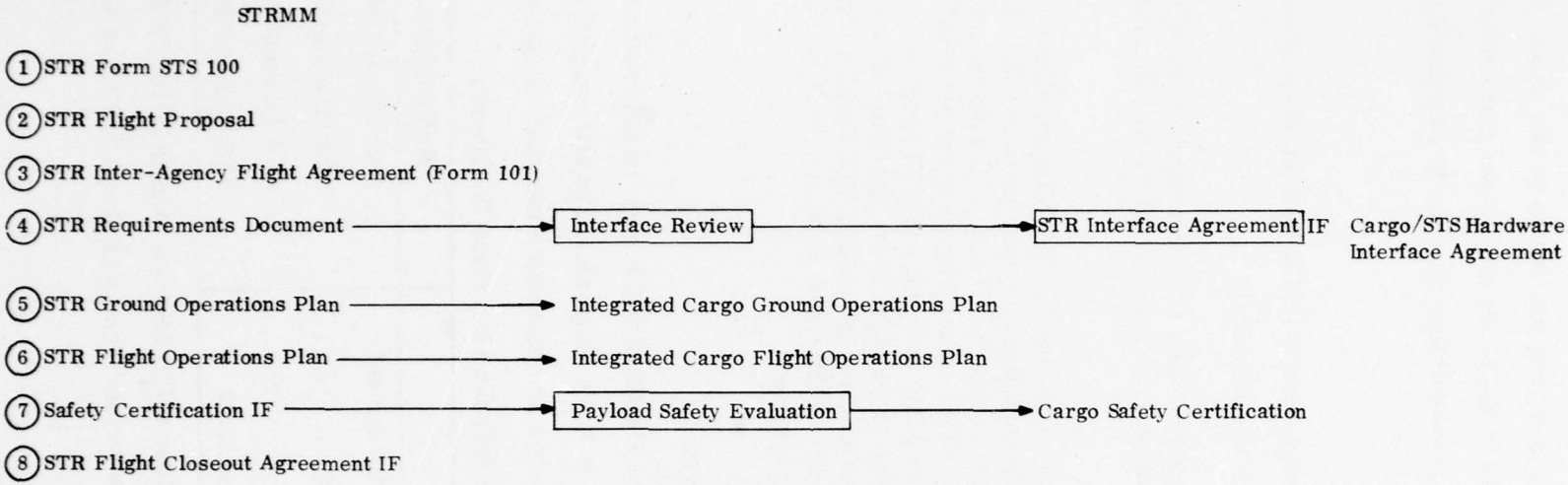
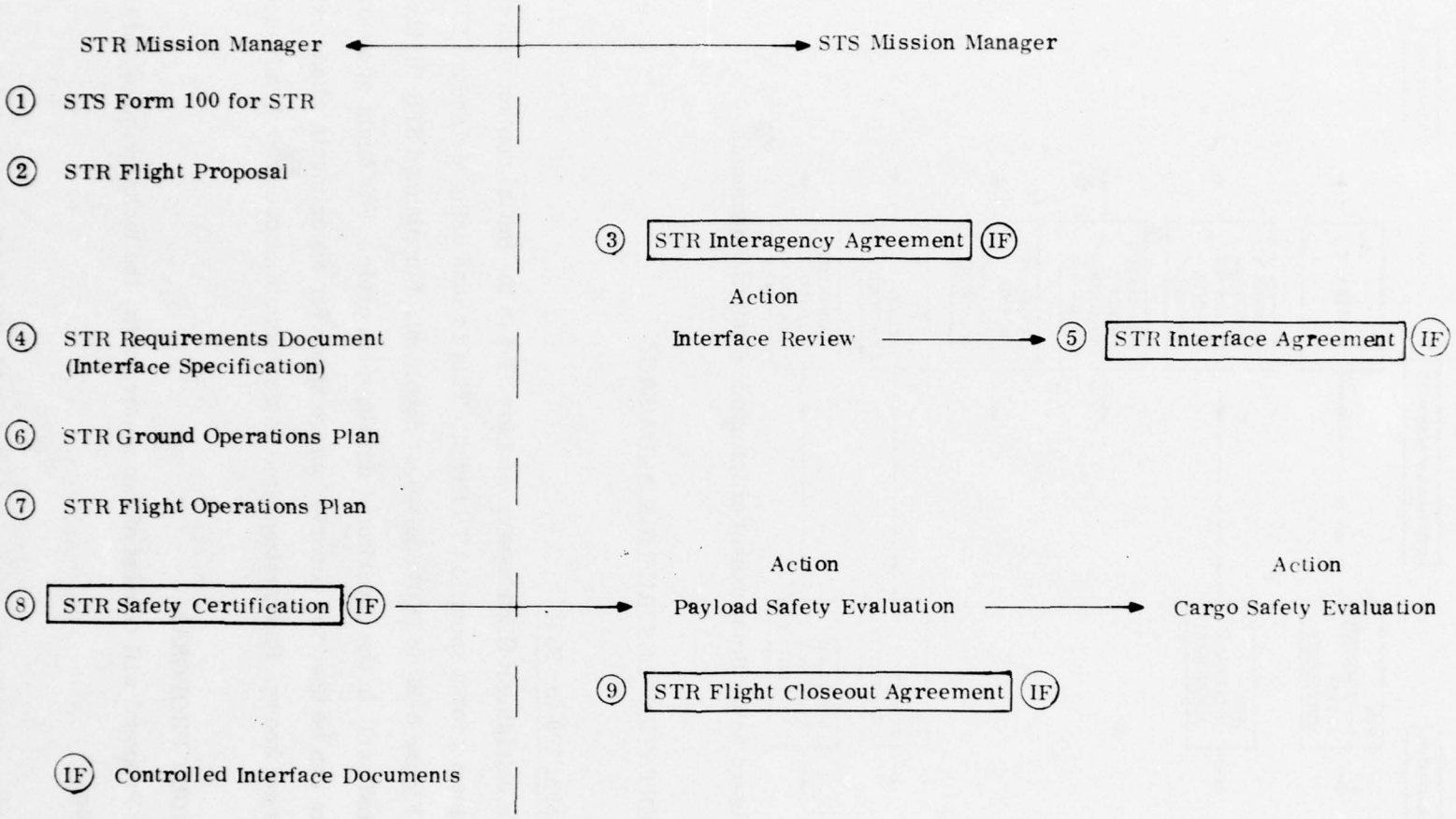


Table 6-3. STR Program Generated Documents



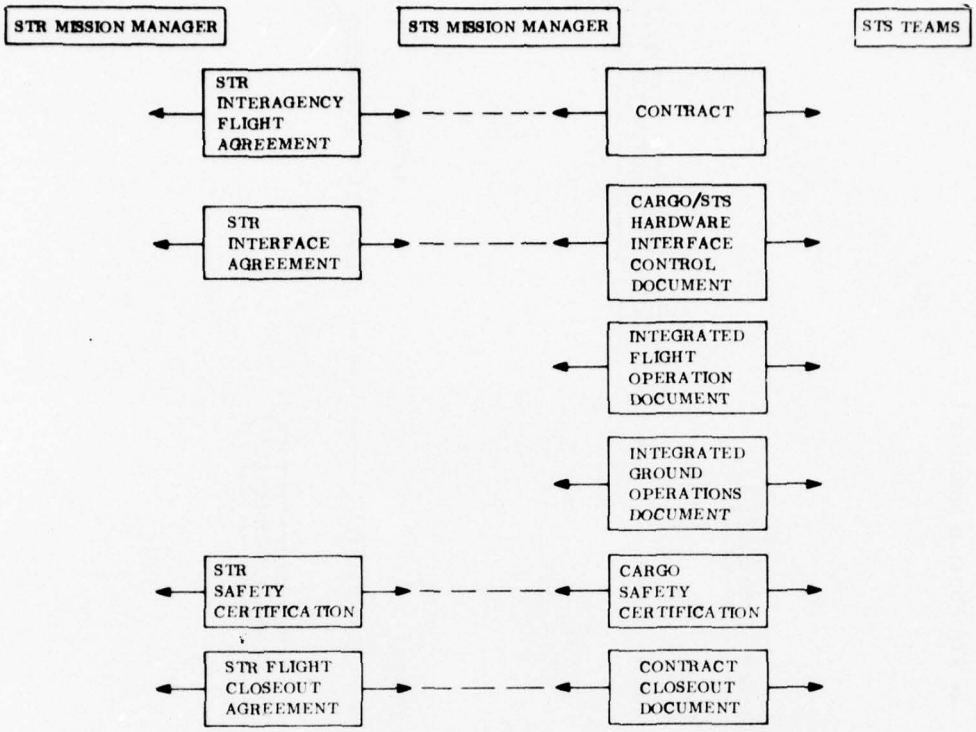


Figure 6-8. Relationship of Interface Control Documents

6.3.3 DOCUMENTATION DESCRIPTION AND USAGE

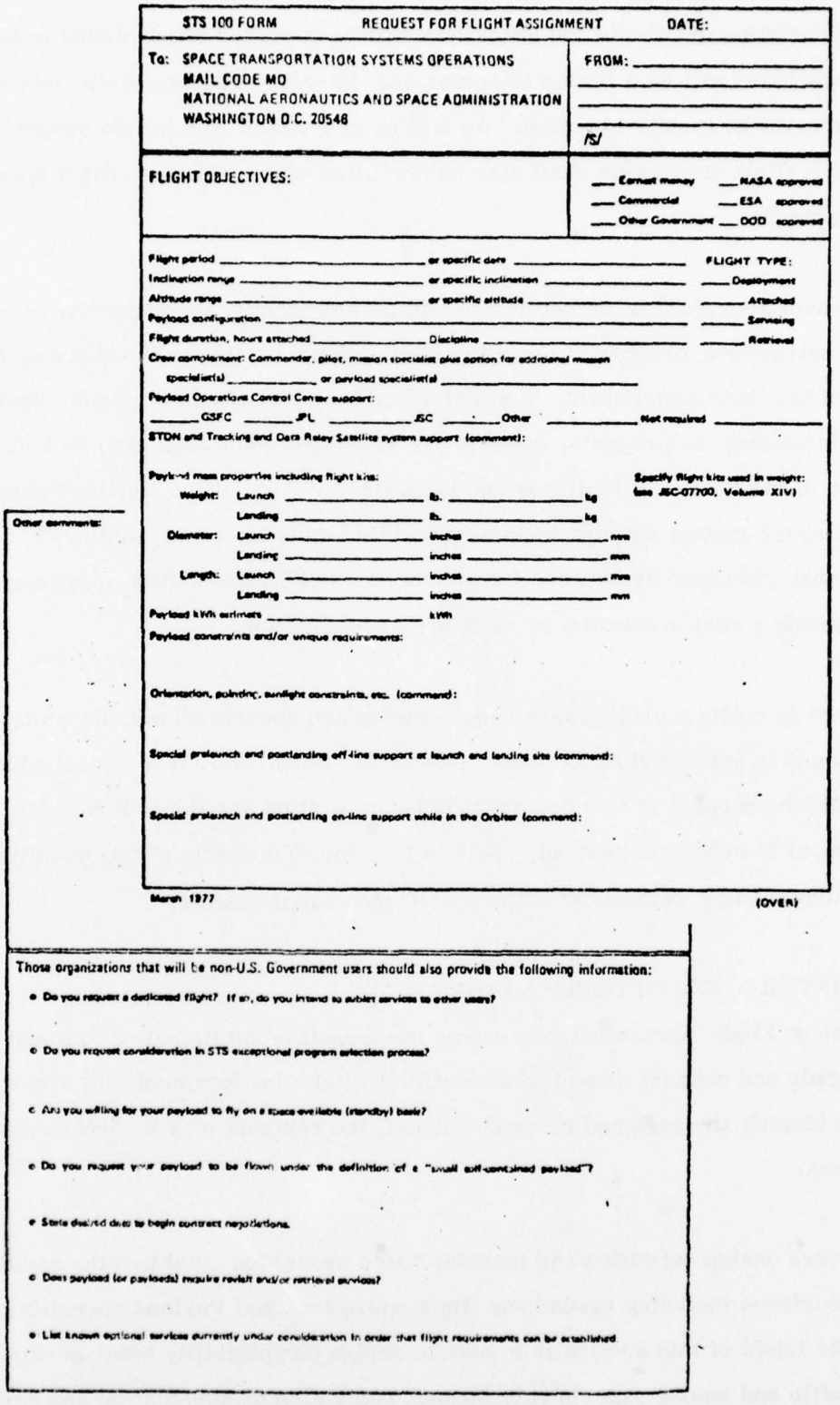
6.3.3.1 STS Form 100 for STR

The first step in obtaining a flight assignment for STR is the initial contact with the STS Mission Manager for a specific STS flight. This is done using a standard STS Form 100 (See Figure 6-9) or DOD equivalent document. For planned STR flights this form is submitted early in the experiment development cycle. For flight of opportunity, this form can be used to "reserve" cargo space for experiments whose exact flight needs are not known. Reservation time is a minimum of one year in advance.

6.3.4 STR FLIGHT PROPOSAL

The STR Flight Proposal will consist of two major parts, the technical plan and the management plan.

The technical plan will contain a statement of the objective of the proposed experiment. As identified in the guidelines for documentation preparation, the experiment standard



STS 100 FORM REQUEST FOR FLIGHT ASSIGNMENT DATE:

To: SPACE TRANSPORTATION SYSTEMS OPERATIONS
MAIL CODE MO
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON D.C. 20548

FROM: _____
/S/

FLIGHT OBJECTIVES:

___ Earnest money ___ NASA approved
___ Commercial ___ ESA approved
___ Other Government ___ DOD approved

Flight period _____ or specific date _____ FLIGHT TYPE:
Inclination range _____ or specific inclination _____ ___ Deployment
Altitude range _____ or specific altitude _____ ___ Attached
Payload configuration _____ ___ Servicing
Flight duration, hours attached _____ Discipline _____ ___ Retrieval
Crew complement: Commander, pilot, mission specialist plus option for additional mission specialist(s) _____ or payload specialist(s) _____
Payload Operations Control Center support:
_____ GSFC _____ JPL _____ JSC _____ Other _____ Not required _____
STDN and Tracking and Data Relay Satellite system support (comment):

Payload mass properties including flight kits: Specify flight kits used in weight: (see JSC-07700, Volume XIV)

Weight:	Launch	_____ lb.	_____ kg
	Landing	_____ lb.	_____ kg
Diameter:	Launch	_____ inches	_____ mm
	Landing	_____ inches	_____ mm
Length:	Launch	_____ inches	_____ mm
	Landing	_____ inches	_____ mm

Payload kWh estimate _____ kWh
Payload constraints and/or unique requirements:

Orientation, pointing, sunlight constraints, etc. (comment):

Special prelaunch and postlanding off-line support at launch and landing site (comment):

Special prelaunch and postlanding on-line support while in the Orbiter (comment):

Other comments:

March 1977 (OVER)

Those organizations that will be non-U.S. Government users should also provide the following information:

- Do you request a dedicated flight? If so, do you intend to sublet services to other users?
- Do you request consideration in STS exceptional program selection process?
- Are you willing for your payload to fly on a space available (standby) basis?
- Do you request your payload to be flown under the definition of a "small self-contained payload"?
- State desired date to begin contract negotiations.
- Does payload (or payloads) require revisit and/or retrieval services?
- List known optional services currently under consideration in order that flight requirements can be established.

Figure 6-9. STS Form 100

fact sheets consisting of fill-ins and checklists will be compiled and included in this book. Also included will be a listing of equipment, interface requirements, consumables and experiment results expected. An outline of Payload Specialists support required for STR flight operations shall also be identified along with post-flight operations requirements.

The management plan shall be structured to outline and identify the organization, including subcontractors, proposed to accomplish the tasks leading to the successful completion of the User Experiment. Schedules and the identity of key people supporting and administering the program, and a program control plan shall also be included. Identification of the type(s) of facilities and laboratories that will be required along with the necessary ground support equipment and its status (new and existing) is also required so that commonality between Users can be established. The management plans will include a cost breakdown by categories and function.

This document is really a preliminary requirements and operations plan document. In some cases it is submitted at the same time as the Form 100. It is conceivable that the requirements part of this document is never written and the STR Requirements Document is submitted instead. This is true for STR flights of opportunity where last minute flight is used for mature STR/payload programs.

6.3.5 STR INTERAGENCY FLIGHT AGREEMENT

The Interagency Flight Agreement represents the formal commitment of STR and STS to perform, supply and support those tasks identified within the document. In order to simplify and identify the assigned responsibilities, the contract may be broken down into four areas.

The first covers design definition and includes those areas that establish the payload-to-Shuttle interfaces including ground and flight operations and Payload Specialist training. The intent of this section is to assure design compatibility between experiment and Shuttle and among experiments through controlled documentation and design review. The second covers integration activities and responsibilities including the

area of payload testing, installation and servicing. This section identifies the operation, support equipment, installation procedures leading to flight readiness and launch. The third area covers flight activities and includes the flight operations responsibilities, procedures and support and data management and disposition. The fourth area covers post flight activities and responsibilities and includes cargo disassembly, STR removal, transportation and handling.

The key documents that apply are identified in the agreement along with the approval and revision procedures that will be followed. The Interagency Flight Agreement also establishes the limits of liability, the binding responsibilities of STR and STS Mission Managers, and the charges applicable to the STR flight.

As previously specified, this will be a standardized document of fill-ins, checklists and fact sheets, particularly applicable to the repetitive nature of STR programs. This document is prepared and updated as required by the STS Mission Manager using inputs from the STR Form 100 and STR Flight Proposal. The Interagency Flight Agreement and the STR Flight Proposal will be similarly formatted so that direct transfer of pages can be accomplished as applicable.

6.3.6 STR INTERFACE AGREEMENT

This document is an updated version of the STR Flight Proposal. It is similar in format but contains detailed and/or more mature data than does the flight proposal.

For planned STR flights, at predetermined periods during the design and development of payload equipment and software, the STR Mission Manager will release to the STS Mission Manager information required to assess the payload and overall cargo design compatibility, and perform the analyses necessary for complete payload/cargo planning and control. These data releases are phased throughout this period to provide a staged release of information to the Mission Manager as equipment design matures. The staging of this data is also set so that information is provided to the Mission Manager in a timely fashion to match his needs.

Some of the information that is transmitted through these releases includes: structural and mechanical interfaces, electrical/electronic interfaces, crew timelines and modes of experiment operation, power requirements, telemetry requirements, command requirements, output data characteristics, general test information, and truth site support operations. This information from STR and all other Users will be combined and used to develop integrated payload/cargo designs and plans for flight and ground operations.

For STR flights of opportunity, all data will be submitted to the STS Mission Manager at one time as soon after the submittal of Form 100 as is possible. The STS Mission Manager will then do whatever analysis and evaluation is necessary to integrate the STR into an existing cargo. It is this effort and the accompanying interfaces, technical exchanges and discourses and physical preparation of the STR/Orbiter interfaces that imposes some limit on the time before launch to periods of one to six months for last minute flight notifications.

The STR Requirements Document is formatted uniformly wherever possible using check lists, fill-ins, and standardized forms in order to facilitate its use as an effective tool for interface information exchange. Following its release and an informal interface design review, a sign-off of the document by the STR Mission Manager and STS Mission Manager will constitute its formal acceptance as the STR Interface Agreement. As a formal document under contractual control, changes will be made in two basic ways. First, the review and acceptance of new data releases will be followed by incorporation of the newly refined data into the Specification. Second, as provided for by the STR Interagency Flight Agreement, formal change notices may be appropriately issued, signed-off, and incorporated as the need arises.

The STR requirements-to-interface agreement concept is illustrated in Figure 6-10. Note: The format of this document will specify a range of maximum or minimum limits for each item as required i.e. (temperature maximum 70° C to 75° C, minimum -5° C to +10° C) to eliminate downstream changes. This is the current approach used in NASA interface agreements which replace the old interface specifications.

6.3.7 GROUND OPERATIONS PLAN

The STR Mission Manager identifies and provides to the STS Mission Manager requirements such as timelines, facilities, support manpower, special environmental conditions, and resources needed for testing, handling and integration of the STR program. This submittal is in a standard format of fill-in lists, check lists and fact sheets. The document is called the STR Ground Operations Plan. It is an input document used to transmit data from the STR Mission Manager to the STS Mission Manager. The document is updated as required.

The Integrated Ground Operations Plan is developed from the needs established by the STR Mission Manager's ground operations plan and from the capabilities shown in the Launch Site Support Manager's (LSSM) ground operations plan for the specific mission. It covers only those items directly affecting both parties.

Having integrated the Ground Operations Plans from STR and other users, the STS Mission Manager identifies and forwards to the Launch Site Support Manager (LSSM) the integrated ground operations requirements needed for checkout, handling and integration of the total flight cargo. The LSSM then integrates these requirements with those of other programs and develops the launch site support plan for that cargo. The LSSM commits resources to the flight to satisfy the cargo requirements. This committal and requirements agreement, when signed by the STS Mission Manager and the LSSM, becomes the Integrated Ground Operations Plan.

6.3.8 FLIGHT OPERATIONS PLAN

A Flight Operations Plan, similar to the ground operations plan, is developed by the STR Mission Manager. Information contained in this document include items such as experiment operations, crew activity and shuttle orbit data, pointing, attitude and maneuvering.

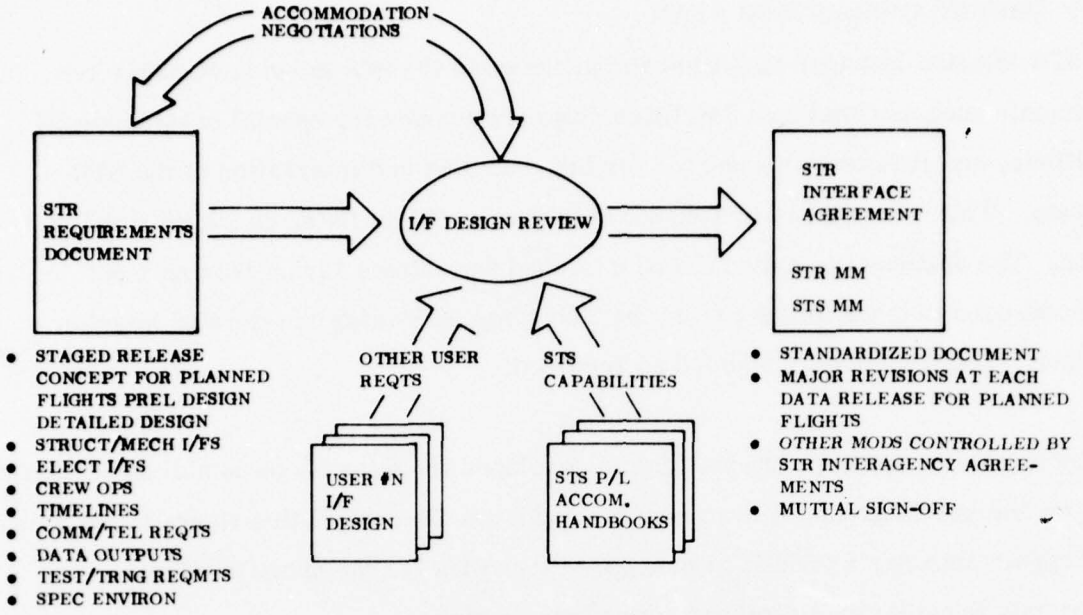


Figure 6-10. STR Interface Agreement

The Flight Operations Plan is submitted to the STS Mission Manager. This submittal is in a standard format of fill-in lists, check lists and fact sheets. The document is called the STR Flight Operations Plan. It is an input document used to transmit data from the STR Mission Manager to the STS Mission Manager. The document is updated as required.

The STS Mission Manager, having combined the flight operations requirements from STR and other Users, forwards to the Flight Operations Mission Coordinator (FOMC) the cargo flight operations plan and support requirements. Information contained within this document are combined with the general STS flight plans for all flights to develop the STS flight operations plan for that particular cargo. The STS FOMC commits resources to the specific flight to satisfy the cargo requirements. This commital and requirements agreement, when signed by the STS Mission Manager and the FOMC becomes the Integrated Flight Operations Plan.

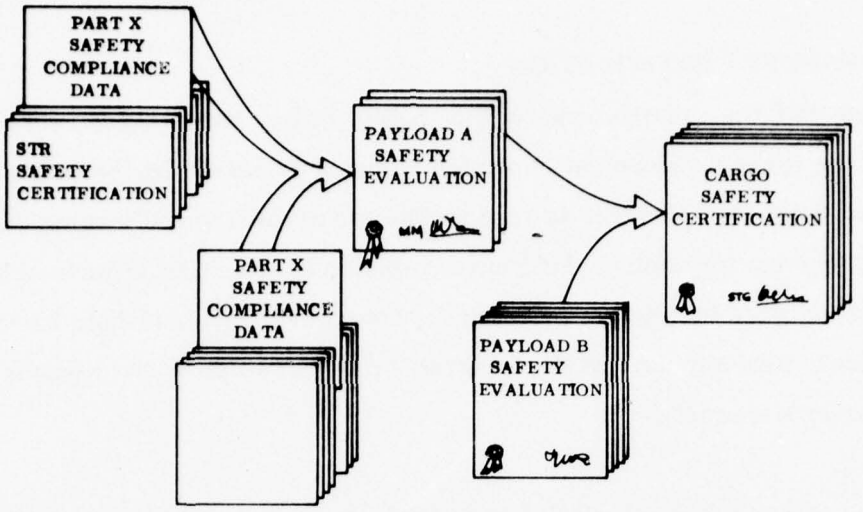
6.3.9 STR SAFETY CERTIFICATION

In accordance with the requirements of the "Safety Policy and Requirements for Payloads Using the Space Transportation System" document issued by the Office of Space Flight, the STS Mission Manager is responsible for preparing and issuing Safety Assessment Reports for each payload in the mission cargo. The reports are composed of data supplied by Users regarding the potential hazards of their hardware and procedures, plus any data on the integrated payload which is the responsibility of the STS Mission Manager.

The STR (and other users) submit Safety Certification reports which include a safety analysis, back up data (much of the interface and requirements data is useable) and a certification that the STR (or other users) payload is safe to fly on the STS.

A series of Safety Assessment Reviews will be held between the STR and other Users and the STR Mission Manager to assure that all potential hazardous materials, procedures, and failures are identified and that safeguards against their propagation are included in the experiment or payload design. An evaluation of safety compliance is issued after the final design is completed, near the end of the analytical integration phase. These evaluations consist first of the STS Mission Manager reviewing and approving the relevant data of the Users, and then the respective STS team representative certifying payloads which make up the cargo. Timing of these activities is a function of the type of STR program, planned or opportunity flight. The flow of activity for planned STR flights is shown in Figure 6-11. For flights of opportunity, shortened cycles of the same activity must be achieved. This can be done by standardization of review procedures and elimination of series activity.

It is assumed that a single safety compliance data package for the cargo results from this first series of formal reviews. This package is then updated as necessary by the Users and STS Mission Manager, and once again reviewed by the STS team representatives just prior to payload shipment to the launch site. Formal signoff of the cargo safety



SAFETY COMPLIANCE DATA

- SAFETY ASSESSMENT REPORT
- APPROVED WAIVERS
- POTENTIALLY HAZARDOUS MATERIALS
- SAFETY-RELATED CONTINGENCIES
- INSPECTION CERTIFICATIONS
- OPERATING PROCEDURES

SAFETY ASSESSMENT REVIEWS

- COMPLIANCE ASSESSMENT
- SAFETY-CRITICAL DATA REVIEW
- SIGN-OFF CERTIFICATES OF SAFETY COMPLIANCE
- APPROVAL OF WAIVERS

Figure 6-11. Safety Certification

data package should be interpreted as approval of the design as it then exists. Changes to hardware and operational procedures which affect the contents of this package must be submitted to the STS Mission Manager and responsible STS personnel for acceptance before implementation. The updating mode represents the path by which STR's for flights of opportunity will be handled.

6.3.10 STR FLIGHT CLOSEOUT AGREEMENT

The STR Flight Closeout Agreement is a post-flight report that records the performance of the two parties as contracted in the Interagency Flight Agreement. It provides necessary and sufficient information to enable these two parties to close out the Agreement. Additional post-flight reports may be issued by either party as required, but the Flight Closeout Agreement is the only contractually required post-flight report.

These documents are required between the STS Mission Manager and STS and between the STS Mission Manager and each User. The former records such things as power levels in the cargo bay, Orbiter altitude/inclination/attitude history, and payload/cargo operating cycles actually run. Normally the document will report discrepancies between what was originally planned and what actually occurred. The latter documents between the STS Mission Manager and each User records such things as levels of Orbiter services provided and Payload Specialist activities. An important part of these documents is the recording of discrepancies between planned and actual provisions. All the parameters which are covered in the Flight Close-out Agreements should have been identified in the initial Interagency Flight Agreement. Parameters that pertain to experiment and Shuttle internal subsystems, as well as those pertaining to scientific data results are not applicable for these documents. They may very well be covered in other User, STS, or even STS Mission Manager generated documents, but these will not be formal interfaces.

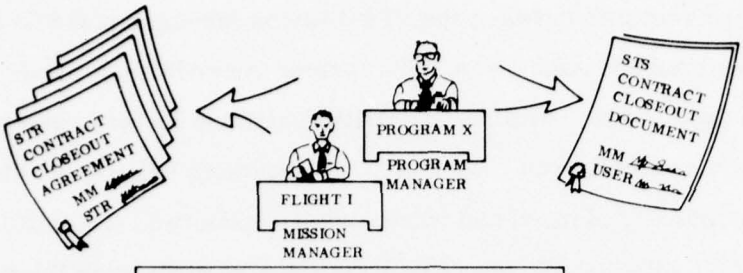
Finally, the User Flight Closeout Agreements will include the settlement of the fiscal areas of charges, penalties and liabilities, past and future. The Flight Closeout document is summarized in Figure 6-12.

6.4 ORGANIZATION AND MANPOWER

An analysis was done of the activities required to implement a full STR program from designation of experiments to fly on STR thru the deintegration of the experiments from the Orbiter after flight to estimate the type and number of personnel required. Guidance for this analysis was provided by recent integration studies and GE experience in the design, development test and flight of space systems and Shuttle integration activites.

6.4.1 ORGANIZATION

The STR is visualized as a continuous program of flights on the STS thru the 1980's decade. A central office headed up by the STR Mission Manager will manage the total STR program. The STR Mission Manager is supported by the organization shown in Figure 6-13. To support him, there are a number of STR Mission Manager deputies,



- FORMAL RECORD OF CONTRACT COMPLIANCE BY MISSION MANAGER, USERS, AND STS
  - SPECIFICATION DEVIATIONS
  - ANOMALIES AFFECTING INTERFACES
  - ACTUAL OPERATIONAL LEVELS, CRITICAL MEASUREMENTS
- NEGOTIATED FISCAL RESPONSIBILITIES
  - REIMLURSABLE USER CHARGES
  - EXCESS USE PENALTIES
  - INTER-PARTY LIABILITIES
- COVERS ALL PHASES OF PAYLOAD ACTIVITIES
  - DESIGN DEFINITION
  - PHYSICAL INTEGRATION
  - FLIGHT AND POST FLIGHT
- DOES NOT INCLUDE
  - SCIENTIFIC DATA
  - ROUTINE/NOMINAL HOUSEKEEPING DATA
  - INTERNAL PARTY PROBLEMS

Figure 6-12. Flight Closeout Agreements

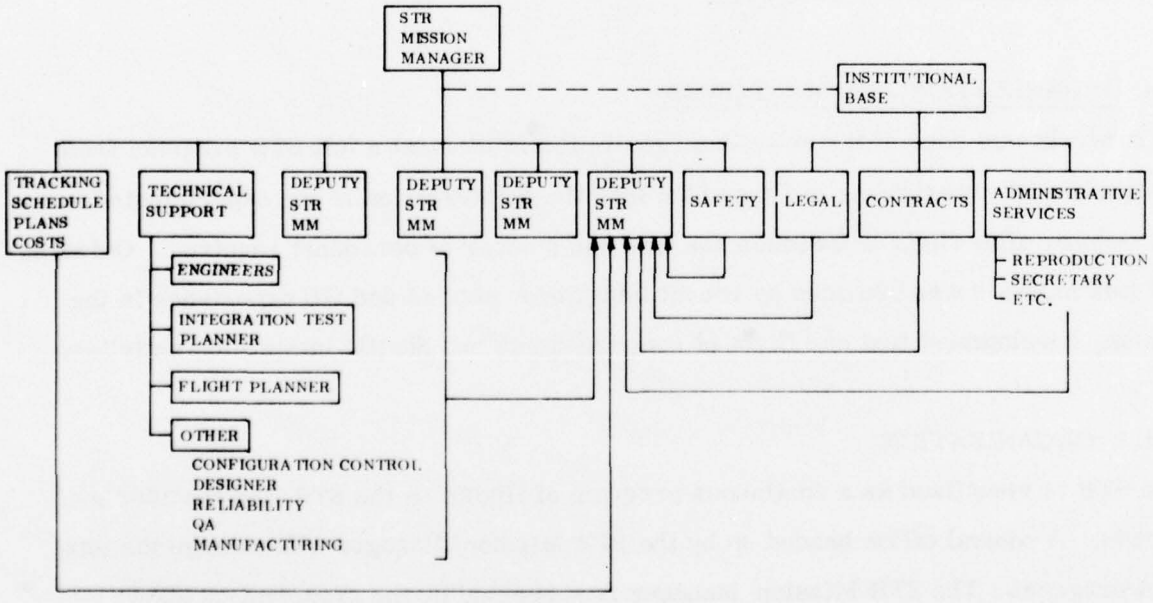


Figure 6-13. Central Office Organization

each of whom will be assigned responsibility for one or more STR programs. The number of programs assigned to each is a function of the particular payload complexity.

Teams comprised of personnel from each of the supporting areas will be set up for each STR program as shown in Figure 6-13. Because of the variation in each STR experiment complexity, many of the individual support personnel will work on more than one program simultaneously. The source of personnel is not identified in this study. They could come from SAMSO, Aerospace, the Payload Integration Contractor (PIC) or a dedicated STR Integration Contractor (STRIC). The number of people working on a given program will vary with the program phase. That number will be affected by the complexity of the experiments. The experiments vary from a few simple hard mounted boxes requiring little crew support i.e. Experiment No. 3, ATLAS, to large payloads requiring a pointing system with attitude control and a high level of payload control, i.e. Experiment No. 9, Teal Ruby.

The tracking, schedule, plans and cost group are responsible for planning and monitoring the schedule and cost performance on all STR programs. It is also responsible for maintaining the STS availability matrix described in paragraph 5.5-1. In addition to the above, they will keep an STR hardware availability matrix to enable future planning for the SPO's with potential payloads.

The technical support staff have as their major concerns, the integration of the experiments and control of interfaces with the STS and other users. They will serve to support the experiment contractors and also to provide information on interfaces with the STS. With the number of STR's and payloads flying on Shuttle, the appointment of a full-time safety representative in the STR Mission Manager organization is essential. His responsibilities include the transmission of safety requirements to experiment contractors, preparation of safety analyses, assessment and evaluation

of individual safety status, and liaison for the STR with the ground and flight STS safety personnel. Administrative staff will be assigned to the STR Mission Manager to be used as required to support the Deputies.

Finally an institutional base will provide essential, but occasionally required services needed by each support team such as legal, contracts, etc.

6.4.2 MANPOWER

Estimates of manpower and organization of STR teams were defined for each of the phases of a typical STR program. The phases are:

STR Program Phases

1. STR Payload Integration
2. STR Integration Planning
3. STR Payload Assembly and Test
4. STR Integration with other Orbiter Payloads
5. STR Installation into Orbiter and Launch Operations
6. STR Flight Operations
7. STR Deintegration

The STR Payload Integration is essentially a design, analysis and test phase similar to integrating a new payload on an existing spacecraft bus. Design, drawings, systems analysis, extension of basic STR analyses done during STR development such as thermal, structural, electrical, etc. are produced and prepared. The aim of this activity is to produce an STR/Payload design definition with backup analyses to allow integration into the Orbiter. This data goes into the interface documentation described in Section 6.3. The organization and manpower to carry out this activity is shown in Figure 6-14. Although the person who runs this group is labeled Mission Manager, he is really one of the deputies.

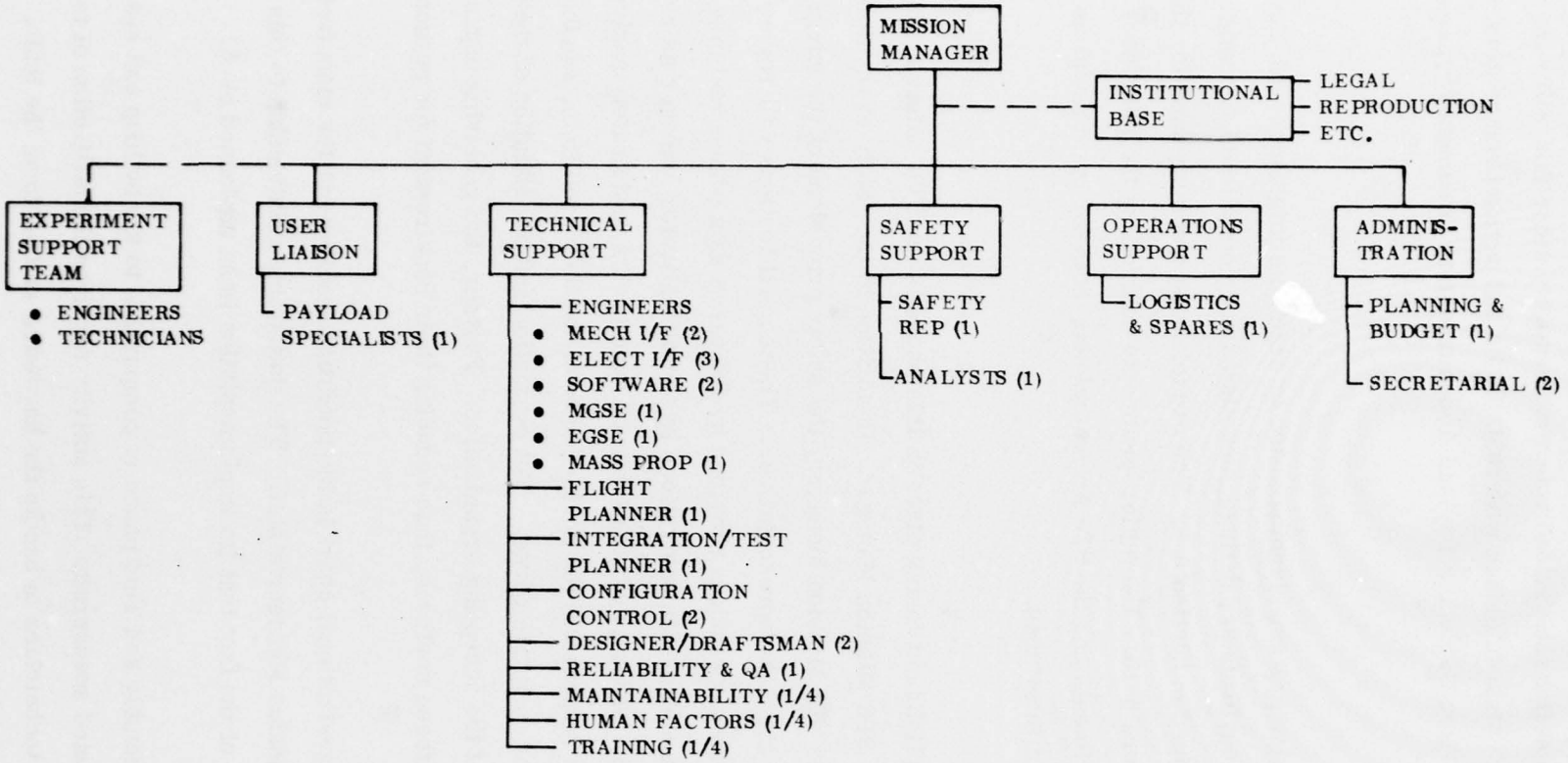


Figure 6-14. Mission Manager Organization STR/Experiment Integration

No attempt is made to define the payload contractor support since this will vary with each experiment. Main interface with DOD SPO's and payload contractors will be through use of a STR User Guide and Payload Accommodation Document prepared by the STR contractor.

The next phase is the Integration Planning phase in which the interfaces with the Orbiter and the other payloads are defined, negotiated and committed by means of the documentation defined in Section 6. 3. The organization and manpower for this task are shown in Figure 6-15. The major documents controlling the interfaces between the cargo experiments and the STS are completed, both for operational as well as hardware equipment interfaces.

Another major accomplishment taking place in this phase is the STR safety evaluation and procedures by the STS Mission Manager. In addition to the safety evaluation of the individual Users by the STS Mission Manager, the safety provisions of the integrated cargo are certified by the STS representatives. These certifications will review the major provisions for safe operations provided by the cargo as a whole, and may also require additional data and/or demonstrations by the STS Mission Manager as required by the STS. The major philosophy being followed here is that basic safety analyses are conducted by the party responsible for performance of the delivered item, whether it is a simple experiment or an entire cargo. But evaluation and certification of these data is the responsibility of the accepting organization. Further, the receiving organization should not reperform these analyses, thus reducing their requirement for personnel.

Also shown on this organizational chart is the potential requirement for manufacturing support in the STR Mission Manager's staff. The dotted line relationship to this function is also indicative of the fact that its implementation is an unplanned need.

The STR/Payload Assembly and Test phase is comparable to the buildup and systems test of an automated spacecraft. This activity requires the utilization of test conductor and testing technicians to handle the hardware and perform the tests.

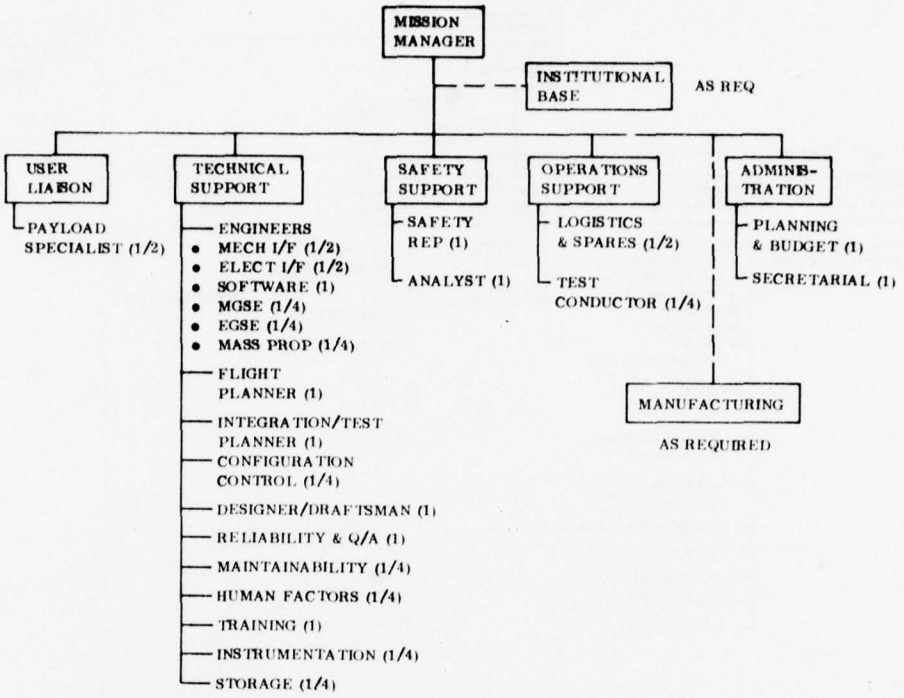


Figure 6-15. Mission Manager Organization STR Integration Planning

These types of people added to the previous organization, make up the staff required to carry out this phase of the STR program. The resultant organization and manpower are shown in Figure 6-16.

The remaining phases of the STR integration into the STS are manned by an organization similar to that shown in Figure 6-16 but with less technical support. A total of approximately 15 people are required to staff these phases.

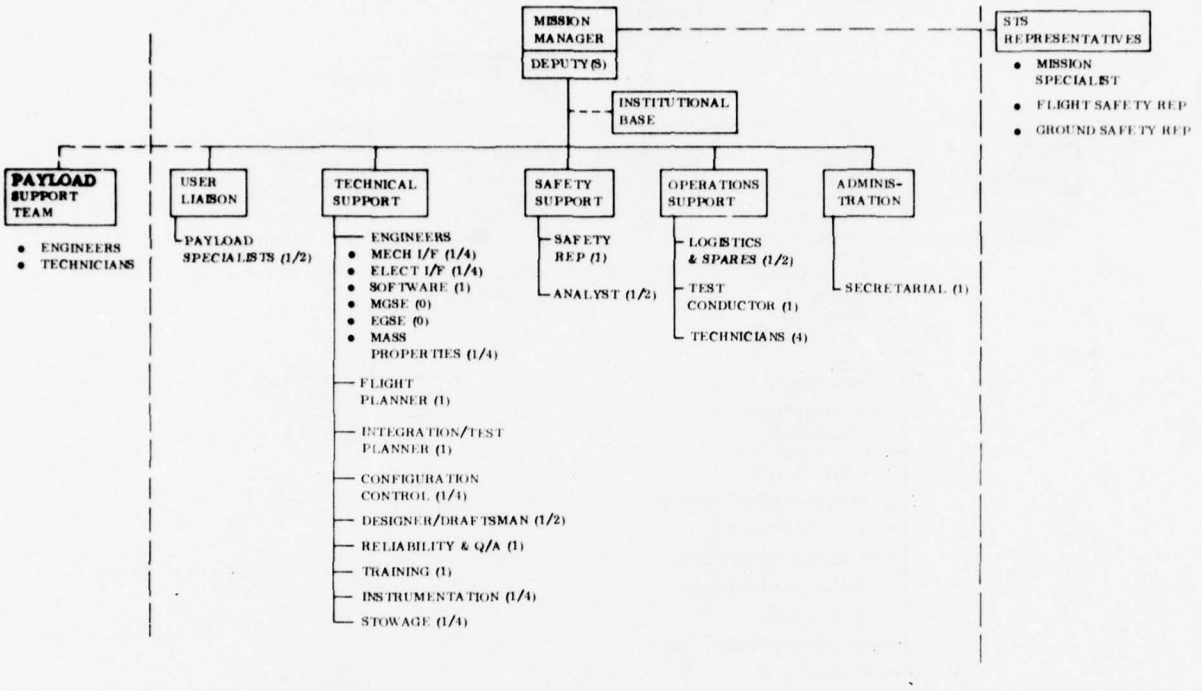


Figure 6-16. Mission Manager Organization STR/ Payload Assembly and Test

SECTION 7
STR COSTS

7.1 GENERAL

The four blocks of costs for an overall 10 year program are shown in Table 7-1. Work Break Down Structures for each of these areas is given in Paragraph 4.7.

Cost for STR development and the acquisition of flight hardware and supporting AGE is shown in paragraph 7.2. The costs of payload integration, integration into the STS and flight are shown in paragraph 7.3 as part of the 10-year Life Cycle Costs.

The ground rules used in this costing exercise are given in Table 7-2. Table 7-3 provides the definition of the baseline 10 year mission. The development schedule shown in Figure 7-1 and the baseline STR one year flight schedule shown in Figure 7-2 were used for the costing exercise. Details of the Life Cycle Costing elements are provided in paragraph 7.3.

The costing approach utilizes individual costs from cost models or scaling known costs for analogous items from previous programs.

During the study the SAMSO and PRICE (Programmed Review of Information for Costing and Evaluation) cost models were exercised for the spacecraft subsystems. This data was backed up with known subsystem cost data for off-the-shelf hardware. The PRICE model is more accurate since the SAMSO model ranges of parameters do not extend to the levels required for the STR hardware. It is aimed specifically at automated spececraft designs.

7.2 STR DEVELOPMENT AND FLIGHT HARDWARE COSTS

7.2.1 INTRODUCTION

The items costed in this section include:

1. STR development and qualification, including ground support equipment and ground station simulator

Table 7-1. Blocks of Costs

1.	STR Development STR Flight Hardware	One Time Comparable to Standard Bus Development
2.	Payload Integration	Each Payload Comparable to S/C Buildup & Test
3.	STS Integration & Flight	Each Payload Flight Integration Into STS
4.	STS Flight	Each Payload

Table 7-2. STR Costing Ground Rules

- 1977 dollars through G & A
- Average complexity weights, lengths and power for the 12 Baseline Experiments
- No experiment costs included
- Integration costs include the payload
- Does not include costs of payload specialists or extra STS services
- One experiment flies on each STR flight.

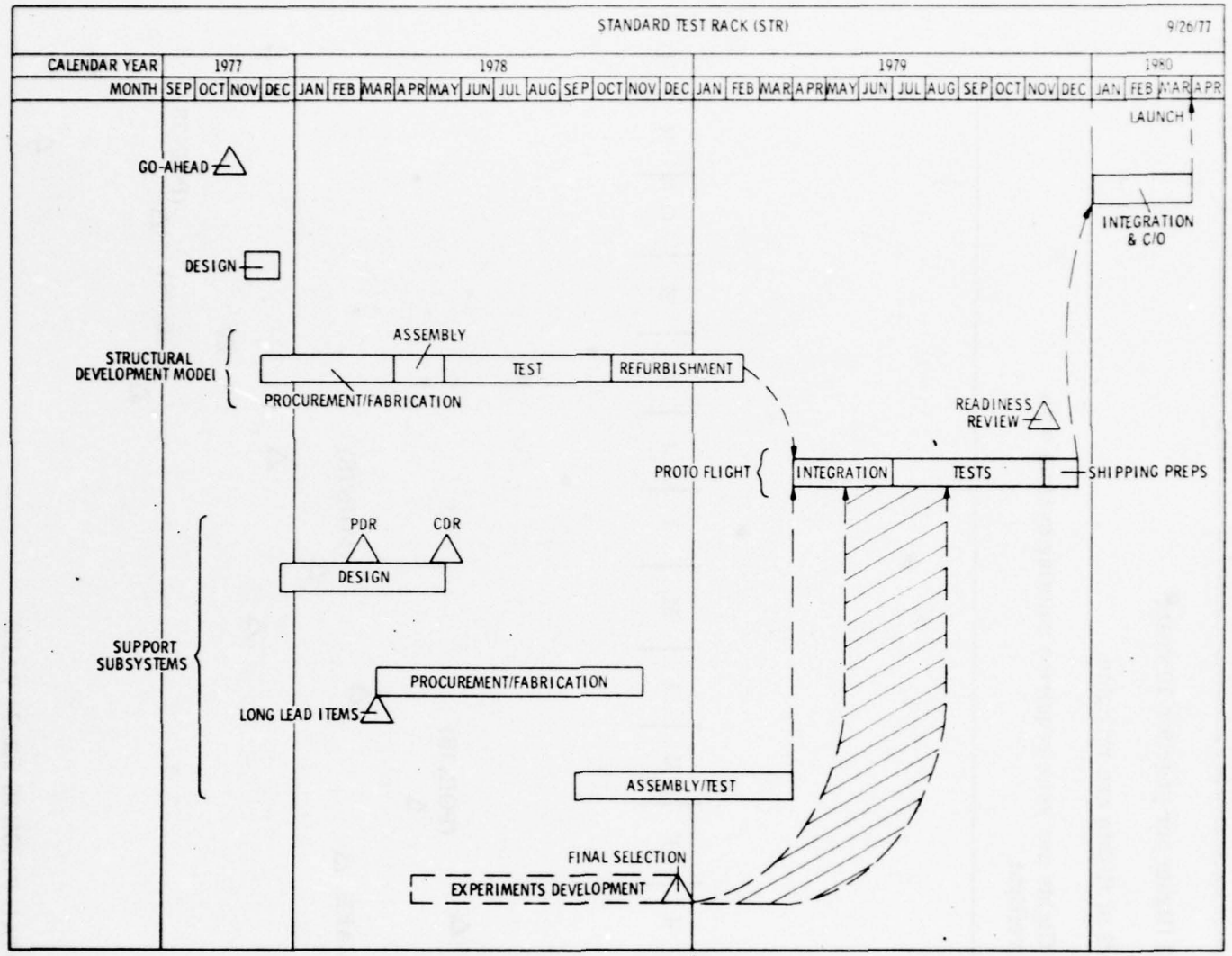


Figure 7-1. Standard Test Rack (STR) Development Schedule

Table 7-3. STR Missions Definition

- 12 flights per year for 10 years
- 1/3 of flights are reflights
- 4 flights per year require a pointing system with 1 reflight

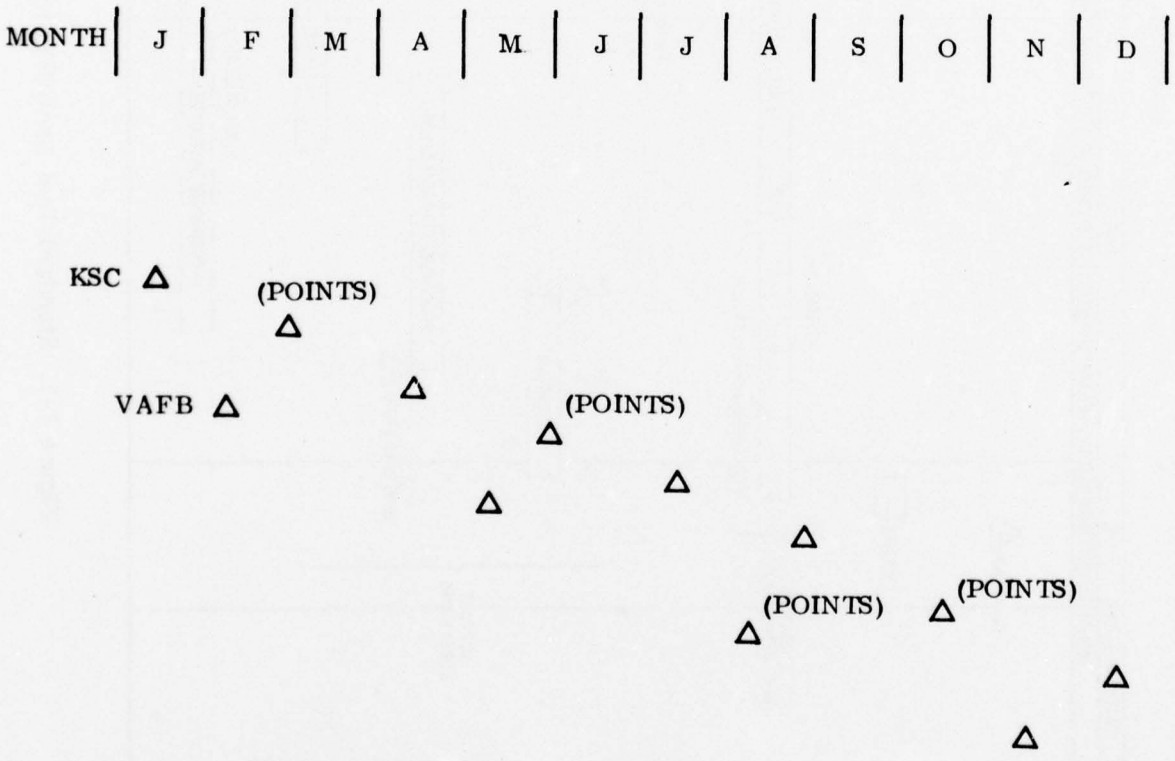


Figure 7-2. Baseline One Year STR Flight Schedule

2. STR flight units and additional ground support equipment and ground station simulator hardware to support the 10 year mission cycle

Based on the STR design, testing requirements and an analysis of the 10 year flight schedule, the hardware in Table 7-4 is required. Maximum use is made of the STR development program hardware for mission support. This is illustrated in Table 7-4.

7.2.2 STR COSTS

A hardware maxtrix for the STR is shown in Table 7-5.

Much of the hardware identified in the spacecraft design configuration described in Section 4 exists and can be used with no modification. Table 7-5 summarizes the design status, supply source and heritage of the spacecraft hardware. Of the 35 elements listed 19 can be used as is, 3 are modified designs and only 13 are new designs.

Table 7-4. Mission Hardware Required

4 STRS
2 KSC
1 VA FB
1 SPARE* At KSC
2 Sets of EAGE
1 KSC*
1 VA FB
2 Sets of MAGE
1 VA FB
1 KSC*
3 POINTS
1 KSC
1 VA FB
1 SPARE* At VA FB
2 Ground Stations
1 VA FB
1 KSC*

*The source of this hardware is the STR development program.

There are few new component designs in the Communication Subsystem, which derives its heritage from prototype equipment used for the BSE and DSCS III Programs as shown in Appendix D. The Encryptor/Decryptor unit is GFE. The tape recorder units are the NASA Standard 10^8 and 10^9 being developed by RCA. The S-band antenna design has been flight-qualified for the Japanese BSE Program.

All of the proposed hardware for the thermal subsystem, except the blanket supports which are spacecraft unique and the change-of-phase hardware, has flown on the Nimbus and Landsat Programs and will require design effort only in the areas of interface, mounting and assembly. The equipment has achieved a high degree of reliability on the flight vehicles in these programs.

The Attitude Control Subsystem hardware has, for the most part, been flight qualified on past military and non-military programs or is NASA Standard for the Multi-Mission Spacecraft (MMS). The Inertial Reference Unit is flying on a current military mission.

The power subsystem utilizes Lithium Sulfur Dioxide cells which are currently used on space programs such as LDEF. Costs were developed for assembling cells into a modular battery as described in paragraph 4.1. The ordnance controller and power controller are similar to GE hardware used on the NIMBUS/LANDSAT and GPS power subsystems respectively.

A summary of the STR non-recurring and recurring costs by WBS item is given in Table 7-6. The MAGE, EAGE and ground stations utilized for this costing are summarized in paragraph 5.6. The test program definition used to define the major development, qualification and acceptance test costs are given in Section 4.6. A simplified STR cost summary is provided by Table 7-7. Table 7-8 provides a "Shopping list" of various STR configuration and subsystem options. This is made possible because of the modular nature of the STR design.

Table 7-5. STR Hardware Design Status Supply Sources Heritage

Item	Status	Qty.	Source	Heritage
Payload Interface Unit	N	1	GE	-
S-Band Antenna	E	3	GE	BSE
Hybrids	E	1	Anaren 10040-4 10040-3	DSCS III
Diplexer	E	1	Wavecom S-704	DSCS III
Switch	E	1	Transc 909C72400	DSCS III
Xmtr/Rcvr	E	2 each	Cubic TR-37	DSCS III
Encrypter	E	2	-	Military & NASA prog
Decrypter	E	2	-	Military & NASA prog
Telemetry Unit	N	1	GE	
Command Decoder	N	1	GE	
Tape Recorder HB	E	2	RCA MASA STD 10^9	MMS
Tape Recorder WB	E	2	RCA NASA STD 10^9	MMS
Inertial Ref. Unit	E	1	Bendix	Other Military Program
Star Sensor Assembly	E	2	Bendix (NASA STD MACS)	
Gyro	E	3	Typical	----
Sun Sensor	E	2	GE	BSE
Computer NSSC-1	E	1	NASA Standard MMS	MMS
Battery	N	1		LDEF, other progs.
Power Controller	N	1	GE	GPS
Power Conditioners	N	2		
Harness	N	2*	GE	Numbus, Landsat, BSE
Ordnance Controller	M	1	GE	Nimbus, Landsat, BSE
Multi-Layer Insulation	M	As Rqd	GE	Nimbus, Landsat, BSE
Heat Pipes	M	As Rqd	GE	BSE
Thermal Coatings	E	As Rqd	Miscellaneous	Nimbus, Landsat, BSE, S-193
Change-of-Phase	M	As Rqd	GE	----
Heaters	E	As Rqd	Tayco	BSE
Thermostats	E	As Rqd	Texas Inst.	BSE
Thermistors	E	As Rqd	Yellow Springs Ind.	BSE
Thermal Tape, Grease	E	As Rqd	Schejdahl, GE	Nimbus, Landsat, BSE
Blanket Supports	N	As Rqd	GE	Nimbus, Landsat, BSE
Structure	N	As Rqd	GE	Modified SEOPS (NASA Version)
Deploy/Retrieval Mech.	E	2	GE	-----
Jettison Mech	E	2		Biosatellite

E	Use as is
M	Modify Design
N	New Design

*Techniques and components are flight qualified. Harness routing is the new design.

Table 7-6. STR Costs By WBS Item

WBS Number	Item	Cost-1977 K$	
		MR	R
1.1.1	STR Integration and Assembly	120	10
1.1.2.1	ACS	1760	1070
1.1.2.1	Points	890	1190
1.1.2.2	TT&C	3400	1730
1.1.2.3	Thermal Control	800	250
1.1.2.4	EP & D	1030	250
1.1.2.5	Structure	1330	230
1.1.3	Payload Envelope Simulator	50	-
1.2.1	Training	-	300
1.3.1	AGE (Points)	300	-
1.3.1	MAGE	830	-
1.3.2	EAGE	350	-
1.3.3	Airborne Support Equipment (As Required)	150	-
1.4	Systems Test & Evaluation	900	265
1.5.1	Systems Engineering	800	200
1.5.2	Project Managment	1220	380
1.6	Data	80	20
1.7	Flight Support	100	-
1.8	Group Communication Command & Control Equipment	1750	-
1.5.2.9	Quality Assurance	400	100
	Operational Site Dedication		
	Total Without Points	14920	4505*
	Total With Points	16260	5695

*Basic STR with no AGE, Ground Station, ASE, or Flight Support.

Table 7-7. STR Cost Summary

Item	Costs, 1977 K$	
	NR	R
Structure	1330	230
Thermal Control	800	250
Power	1030	250
ACS	1760	1070
Communications	3400	1730
Subsystem Total	8320	3530
Program Management	1300	400
Systems Engineering	900	200
Systems Test	1070	-
Acceptance Test	-	275
Reliability & Product Assurance	400	100
Total	3670	975
STR TOTAL	11990	4505
Points	890	1190*
STR with Points	12880	5695
Points AGE	300	-
MAGE	830	-
EAGE	350	-
Ground Station	1750	-
AGE Total	3230	-
ASE	150	
STR TOTAL SYSTEM	16260	

*Cost of a second unit. These item are not on a 1 to 1 basis with STR's

Table 7-8. Potential STR Configuration and Costs

CONFIGURATION	COST, 1977 M $
FULL STR CAPABILITY	5.70
STR WITHOUT POINTING	4.51
STRUCTURE, THERMAL CONTROL, POWER, COMMUNICATION	3.14
STRUCTURE, THERMAL CONTROL, POWER, DATA STORAGE	2.10
STRUCTURE, THERMAL CONTROL, POWER	0.73
STRUCTURE, THERMAL CONTROL	0.48
STRUCTURE ONLY	0.23

The first POINTS unit cost is the utilizing existing hardware for the gimbal system. Recurring costs are for a production unit redesigned to utilize aluminum instead of beryllium.

The cost of a flight console used on the aft flight deck of the Orbiter is shown as typical cost.

The non-recurring costs produce a full set of usable STR hardware and support equipment. The recurring cost is as shown for STR. The cost of an additional set of MAGE, EAGE and ground station is $2500. A set of this hardware is required for each launch site.

The PRICE estimates for the non-recurring and unit recurring costs of the standard subsystems for the Sortie spacecraft were made at the subsystem level and are shown in Table 7-9.

As related earlier, available subsystem costs were assembled and compared with the results of the PRICE model costs.

PRICE methodology provides reliable estimates of system acquisition costs (Development and Production) during the conceptual phase of a system development program. General Electric has used PRICE previously to assist in developing costs for a number of programs, including DSCS III and TDRSS. With the proper choice of descriptors and the use of historical data to calibrate or tune its algorithms, PRICE is a useful tool for cost estimation.

It allows the user to specify the amount of electrical and structural design remaining to be performed on an item before drawings can be released to manufacturing. Because of the maturity of the designs (see Appendix D) chosen for the subsystem, the factors specifying the remaining design were appropriately selected to be quite low. In general, the amount of structural design (or redesign) remaining was set higher for each subsystem than the electrical design remaining because of new packaging requirements.

Table 7-9. Comparison of STR Costs and Price Costs

Subsystem	Costs, 1977 M$			
	Non-Recurring		Recurring	
	Price	STR	Price	STR
Communications	2.05	2.64	1.2	1.73
ACS	1.51	1.76	0.57	1.07
Power	0.44	1.03	0.17	0.25
Thermal	0.86	0.80	0.40	0.25
Structure	1.84	1.33	0.63	0.23
Total	6.70	7.56	2.97	3.35
Systems Engineering	0.18	0.90	0	0.2
Integration & Test	1.14	1.07	0-33	0.28
Program Management	1.22	1.70	0.83	0.5
Total	2.54	3.67	1.16	.98

The complexity of the design effort can be input to PRICE in terms of an engineering complexity factor. This factor ranges from 0.3 to 2.9 which respectively characterize simple redesign and difficult advances in the state-of-art. Again, becuase of the maturity of the subsystem designs, this factor was selected to represent routine redesign, use as is, or new design in the few cases when necessary.

The remaining factors required as inputs to PRICE were derived from experience on several spacecraft programs.

A comparison of these costs to PRICE cost results for major subsystems is given in Table 7-9. The comparison indicates relatively good agreement. As has happened in the past, using early design definition for PRICE, the costs below the system level varies but the net system level agreement is good.

7.3 LIFE CYCLE COST (LCC)

The STR LCC is determined using the 10-year baseline mission and the costing ground rules defined in paragraph 7-1.

The cost elements included in the STR are listed in Table 7-10. These elements are discussed in the following sections. The basis for LCC costs are shown and an LCC summary is presented.

7.3.1 STR DEVELOPMENT

The STR development costs (excluding POINTS) based on the data in paragraph 7.2 are $16260K. These costs include the following:

1. Design
2. Fabrication
3. Test (Development and qualification)
4. MAGE (useable for 10 year mission)
5. EAGE (useable for 10 year mission)

Table 7-10. Cost Items Included In the STR LCC

- STR Development
  - Design
  - Test (Engineering and Qualifications)
  - MAGE
  - EAGE
  - Ground Station
  - ASE
- STR Flight Hardware
- STR Expendable Hardware Batteries
- STR Spares
- Pointing System Hardware
  - MAGE
  - EAGE
- Payload Integration
- Integration of STR into STS
  - Analytical & Design Assembly and Test
- Flight Operations Support
- STS Flight Costs
- Maintenance & Repair
- Logistics Transport by Airforce
- Training

6. Ground Station Simulator (usable for 10 year mission)

7. STR Unit (Qualification; useable for flight)

8. Basic Software

9. Basic ASE

7.3.2 STR FLIGHT UNITS

The cost of the three additional flight units for the complement of four units required to carry out the baseline mission is $13,515K. This does not include the cost of batteries which must be provided and replaced for each STR flight.

7.3.3 EXPENDABLE HARDWARE, BATTERIES

Based on the average power and energy required for the 12 baseline experiments, approximately 3 to 4 batteries per flight are required. A total of 420 batteries is needed for 120 flights. At an estimated cost of $1100 per battery, total battery cost is $462K.

7.3.4 POINTING SYSTEM

The POINTS system has been assumed to be the pointing system for this cost study. The cost of the unit to be utilized with the development program is $890K (assumed use of existing beryllium gimball system with mounting modifications, 3 gyros, drive electronics and encoders). A complement of ground support equipment for the STR qualification program is $300K including additions for multiple configuration mounting on STR. Additional sets cost $300K. This is based on GE data and pointing system STS integration studies.

The total mission costs for POINTS is:

2 POINTS Systems (modified design production units)	$2380K
1 Sets of support equipment	$ 300K
	$2680K

This hardware is in addition to the development program.

7.3.5 SUPPORT EQUIPMENT

The MAGE, EAGE and Ground Stations elements have been costed by comparison to similar equipment from other GE programs such as Nimbus, Landsat, BSE and DSCS III.

No attempt was made to determine the cost of the ASE. This consists of the console mounted on the aft flight deck for use by the Orbiter crew for STR operation and control. The cables required to connect this console and the C&W system to the STR are also part of the ASE. The cost of this hardware was assumed to be $150 per experiment and to require $10K of update hardware for each experiment.

7.3.6 PAYLOAD INTEGRATION

The cost of payload integration is based on the following groundrules:

1. One payload only on each STR.
2. One third of the flights are reflights, requiring only a checkout of the integrated STR/payload prior to reflight.

The major categories of activity are as follows:

1. Design. Configuration definition and top level installation and assembly drawings for the payload.
2. Analysis. Update of the basic analyses done for an "envelope" payload during the development program. These include thermal, structural, dynamic, acoustic, mass properties, EMC, RF, safety, etc.
3. Integration Test. The tests consist ambient condition electrical compatibility, interface verification and performance. This activity takes place at the dedicated STR preparation area at WTR and ETR.
4. Support Activities. Systems engineering, project management, product assurance, manufacturing, etc.

A baseline cycle was evaluated and it was found that approximately 140 equivalent man-months (approximately 20 people spread over a 7 month period) was required for the non-test activities and 20 man-months was required for the 1 month test period.

Another cost is the design and fabrication of payload unique thermal control elements such as insulation blankets, canisters, heater systems, etc. This cost is estimated at $20K.

An additional cost is the development of software by updating the basic software from the STR development program. This cost is estimated at $50K.

During the payload integration cycle, testing at the payload contractor's plant using an STR simulator will also be done. The total cost of one cycle of payload integration is $1260K.

7.3.7 STR INTEGRATION INTO STS

The integration of an STR/payload assembly into the STS consists of 2 parts, the integration planning and the physical integration in the Orbiter.

The integration planning consists of the interface definition, interface documentation, interface liaison with STS mission manager and STS operation teams. All the other activities required to support these efforts such as STR mission, performance and safety analysis and management are part of the integration planning. An analysis of these activities indicates that 60 man months of effort is required. In addition significant travel and living documentation and computer support costs will be accrued. This activity can occur over a period of several years to as little as 6 months of time. Only the actual working times have been included in the costing.

The physical integration into the STS includes STR activity and support of STS test teams in the following areas:

1. Interface verification testing in the CITE or SIVE.
2. Installation in the Orbiter in the horizontal position.
3. STR activities in the PPR and the PCR. Include STR/payload servicing and testing, installation of the STR in the PGHM, and installation into the Orbiter as required.
4. Post-installation interface verification testing.

An estimate for a typical payload for these activities is approximately 750 man-hours covering packing, moving, unpacking, servicing, installation and test. A total cost for the STS integration is $400K.

7.3.8 MAINTENANCE AND REPAIR

The maintenance and repair of STR covers the following elements:

1. STR
2. Ground equipment
3. POINTS

The STR is designed to require minimal maintenance and one overhaul in 10 years. Maintenance consists primarily of replacing high failure rate items, (from the spares inventory in paragraph 7.3.8), refurbishment and replacement of insulation blankets, cleanup and inspection of critical joints, trunions, fasteners and parts where relative motion is required. Cost is estimated at 2 people on a continuing basis and an overhaul cycle of 2 months requiring 30 people. The total cost of STR maintenance and repair is $1000K for the day to day maintenance and $250K for the overhaul. The cost of maintenance of the POINTS system is estimated on the same basis as the STR. The POINTS maintenance cost is estimated to be $200K over the 10 year period.

The ground support equipment maintenance and repair primarily involves the electrical equipment of the EAGE and the Ground Station. Estimates of these costs from the Nimbus and Landsat programs, are $36K per year. Using $50K and considering two locations, WTR and ETR, the total 10 year cost is $1000K.

7.3.9 SPARES

The spares for the STR are estimated as follows:

4 sets of thermal control elements	$ 450K
3 tape recorders (high failure rate items)	$ 600K
30% STR spares (selected item of electrical equipment)	$2710K
POINTS (10%)	$ 300K
TOTAL	$4060K

7.3.10 LOGISTICS

Transport of major elements is assumed to be by Air Force carrier airplanes. The move of two full sets of STR and support equipment and two additional STRs costs $35000. Estimates of rates for transferring equipment and spares is estimated at $10K per year and $20K for moving 4 STRs to and from the factory repair site. The total logistics costs are $15K per year.

7.3.11 TRAINING

It is assumed that no dedicated payload specialists are required. The training of a payload specialist for each payload operation is estimated to be 4 to 6 weeks, an average of 200 hours for each payload. The training for STR operation is assumed to occur only 10 times and is expected to be about the same as for a payload. The total training time in hours is 18000 hours, approximately $600K.

7.3.12 STS FLIGHT COSTS

The STS flight costs are based on the 1975 DOD charges of $12.2M for a dedicated flight updated to 1977. It is also assumed that all flights will pay an average premium of 1.13 for flight notification between 9-14 months as an average. The calculation of a dedicated flight cost is shown below.

1. $12.2M in 1975 dollars (Reference 6).
2. Inflation factor 1.19 (2 years) for 1977.
3. Short term flight contract premium 1.13.
4. 1977 STS DOD Flight Cost is $16.4M.

The average Orbiter flight capability is conservatively calculated by assuming that 1 out of 3 flights will be a spacelab flight with a weight limit of 32000 pounds (landing load limit) and 2 out of 3 flights will be non-spacelab flights with an average capability lower than 65000 pounds. For these flights, orbit altitudes and inclinations flown from ETR and WTR, a value of load capability of 40000 pounds is estimated. The average flight capability is then $2/3 \times 40,000 + 1/3 \times 32,000 = 37,300\#$.

The average weight of the 12 baseline experiments and STR combinations is 4000 pounds and length is four feet. It is assumed that 2 flights each year will be at a length of 8 feet based on baseline experiment data.

For a payload that will not require an entire flight capability and that can share the cargo bay with others, the cost to the user will be a fraction of the dedicated-flight price, calculated as follows:

1. The payload weight is divided by the Shuttle payload weight capability at the desired inclination and altitude to find the weight load factor.

2. The payload length is divided by the length of the cargo bay, 60 feet (18.29 meters), to find the length load factor.

3. The load factor (length or weight, whichever is greater) is divided by 0.75 to determine the cost factor.

The STS 10 year flight costs for STR are:

$$C = \frac{4000}{37300} \times \frac{\$16.4M}{0.75} \times 100 \text{ Flights} = \$234M$$

$$C = \frac{8}{60} \times \frac{\$16.4M}{0.75} \times 20 \text{ Flights} = \$ 58M$$

Total Flight Costs $292M

7.3.13 LCC COST SUMMARY

A summary of the LCC costs is contained in Table 7-11 for 120 STR flights with single payloads over a 10 year period. The total cost of $504.047 M is conservative and is based on an average STR/payload weight of 4000 pounds with an average payload weight of 840 pounds. For a more efficient flight with payloads of higher weight, the cost to fly a given total payload weight over 10 years is significantly reduced. As an example, if the same total 10 year payload weight were flown with an average payload weight of 2000 pounds per flight only 20 flights would be required and the total LCC would be $264.643 M.

The cost per flight is $4200K, over half of which is STS flight costs. The STR cost per flight including amortization of all hardware costs is $354K per flight. The experiment integration costs are $1362K per flight as shown in Table 7-12. The STR hardware life cycle costs are obviously a small part of the total mission costs for the STR program.

Table 7-11. LCC Cost Summary

	Cost, 1977 K $
• STR (Non-recurring)	16260
1 STR (Usable for Flight)	
1 Set MAGE	
1 Set EAGE	
1 Ground Station	
1 POINTS (Initial Unit)	
1 POINTS AGE	
• 3 STRs	13515
2 ETR	
1 WTR	
1 SPARE (Initial Unit) at WTR	
• Batteries	462
• Pointing System (POINTS)	2380
3 Systems (1 is Initial Unit)	
• Support Equipment	2500
2 Sets EAGE (1 is Initial Set)	200
2 Sets MAGE (1 is Initial Set)	750
2 Ground Stations	1250
2 Set POINTS AGE (1 is Initial Set)	300
• Airborne Support Equipment	950
1 Console, 80 Updates	
• Experiment Integration	104220
80 Full Cycles	100800
40 Retest Only	520
POINTS Integration (40 Cycles)	2900
• STR/STS Integration	58500
Integration Management (Mission Manager and Staff)	5200
Integration Planning	48000
Integration into Orbiter	4800
Site Activation	500
• Maintenance and Repair	2450
STR	1250
POINTS	200
Ground Support Equipment	1000
• SPARES	4060
STR	3460
POINTS	300
Ground Support Equipment	300
• Logistics	150
• Training	600
• Flight Costs	298000
STS Flight (120 Flights)	292000
Flight Support	6000
Total LCC - 10 Years	504047

Table 7-12. STR Costs Per Flight In 1977 K $

Total LCC	$504047
Total Per Flight	4200
Hardware	354
Integration	1362
Flight	2484

A cost sensitivity analysis was performed where the number of flights, integration costs, hardware costs, multiple payloads and number of launch preparation sites were varied. The results of these analyses are given in Table 7-13.

Figure 7-3 shows a typical cost spread for 10 years assuming 12 flights per year. Since the launch costs are a major part of the total costs, the maximum per year expenditure is almost linear with the number of flights.

Table 7-13. Cost Sensitivity Analysis Results

Costs 1977 K $	Baseline	Decreased Number of Flights (One-Half)	Decreased Integration Costs (One-Half)	Increased STR Hardware Costs (Double)	Multiple Payload Costs (Two Per Flight)	Single STR Preparation Site
• Flight	4200	4507	3563	4412	4909	4200
Hardware	354	614	354	566	354	331
Integration	1362	1409	725	1362	1778	1385
Flight	2484	2484	2484	2484	2775	2484

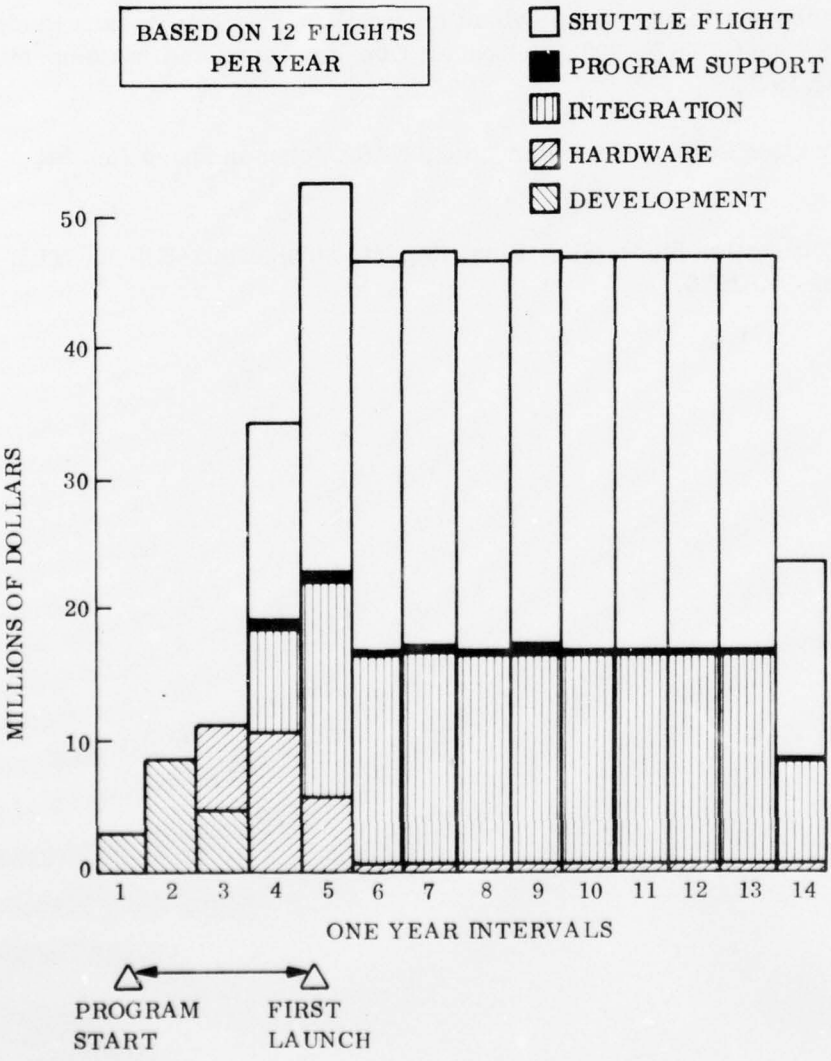


Figure 7-3. STR Life Cycle Costs - 10 Year Spread

AD-A064 773 GENERAL ELECTRIC CO PHILADELPHIA PA SPACE DIV F/G 22/2
STANDARD TEST RACK EXPLOITATION STUDY.(U)
DEC 77 W P ENGLE F04701-77-C-0116
UNCLASSIFIED SAMSO-TR-77-190 NL

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A064773



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5. DOD Space Transportation System Payload Integration Interface Support Study, Final Report, SAMSO-TR-76-103, McDonnell Douglas Astronautics Company, dated December 1976.

6. Space Transportation System User Handbook, NASA Johnson Space Center, June 1977.

7. STP Spacelab Utilization Study (SUS) Final Report, 30297-6001-RU-00, TRW, dated November 15, 1976.

APPENDIX A

CHANGES IN COMMUNICATION SYSTEM FOR STDN COMPATIBILITY

APPENDIX A

CHANGES IN COMMUNICATION SYSTEM FOR STDN COMPATIBILITY

SUMMARY

A review of the reference (1) document was made and it is concluded that the NASA MMS C&DH module with appropriate (substantial) modifications may be utilized as a standard Communications Subsystem (CS) for such satellite programs as Teal Ruby, SIRE, and Lasercom. A description of the changes to the MMS module necessary to perform the CS requirements is presented herein. In order to make a SGLS compatible system compatible with STDN, reverse the changes as those described herein.

REQUIREMENTS

Table A-1 summarizes the communications and data handling requirements for the MMS C&DH module and the CS. Major differences include frequencies modulation, formats polarization, and security equipment. In order to use the MMS module for the CS the following changes are required:

1. Replace the STDN Transponder with an SGLS Transponder.
2. Replace the PSK demodulator with an SGLS signal conditioner (includes FSK demodulator)
3. The signal conditioner and the command decoder must be made compatible by modifying the signal conditioner output, modifying the command decoder input, or by adding a suitable piece of equipment between the two
4. Redesign the MMS omni-antenna.

Further details on the interchanging MMS/CS communications components are summarized in Table A-2 and its footnotes. While the differences between the two systems is substantial, it is possible that components could be interchanged with minimal impact. However, a compatibility interface effort is required early in the program. This effort, primarily consisting of meetings with NASA and/or the MMS C&DII module vendor, would be to review subsystem requirements and electrical/mechanical interfaces, and to identify the details of the hardware that must be deleted, modified, and/or added.

Table A-1. Communications and Data Handling Requirements

	MMS	CS
Telemetry/Command Band	S-Band	S-Band
Tracking System Compatibility	NASA STDN	AF SCF
Uplink Frequency MHZ	2025-2120	1750-1850
Uplink Subcarrier Modulation	PSK	Tenary FSK With AM
Command Format	96 bits	TBD
Command Bit Rate	2k, 1k, 125 bps	2 Kbps maximum
Downlink Frequency	2200-2300	2200-2300
Telemetry Format		
Word Length	8 bits	8 bits
Minor Frame Length	128 words	variable
Major Frame Length	128 minor frames	variable
Maximum Telemetry Bit Rate	256 Kbps	256 Kbps
Power output	1.7, 3.5 or 7.1w	2 w
COMSEC	Not Available	Available
Antenna Polarization	1 with RHCP and 1 with LHCP	RHCP
Memory Size	16K words x 18 bits/word	TBD
Tape Recorder Capacity	$9x10^8$ bits	$3x10^8$ bits
Payload Data Rate	2 Mbps	1.024 Mbps

Following this effort, trades are necessary to determine the preferred approach; use of a modified MMS module or a standard CS approach such as that described in this report (paragraph 4.2).

It should be noted that although the NASA MMS module is capable of transmitting payload data up to 2 Mbps, the SGLS transponder normally used for telemetry, command, and tracking can only transmit data up to 256 Kbps. In order to transmit data at rates up to 1 Mbps, a separate transmitter (carrier II) is required as described in paragraph 4.2. Therefore for operation with the SCF, a separate payload data channel

Table A-2. C&DH Changes Required to Run CS Mission on the MMS

MMS Equipment	Changes to MMS for CS Compatibility	CS Equipment
	Payload module[a]	Antenna
	Payload module[a]	Hybrid
Diplexer		Diplexer
Transponder	Replace[b]	Receiver
		Transmitter
	Payload module[a]	Switch
PSK demodulator	Replace[c]	Dual Signal conditioner
	Add[d]	Decryption unit
Central command decoder[e]	Modify or replace[f] (software change)	Command decoder
Pre-modulation processor	Replace[g]	Dual baseband unit
	Add (if necessary)[h]	Encryption unit
Tape recorder	Usable[i]	Tape recorder
Data bus controller[e]		
Clock and format generator[e]	Usable[j]	Bus controller[p] (data formatter)
Standard computer interface		
Computer	Usable[k]	Computer[p]
Remote unit	Usable[l]	Data interface unit[p]
Harness and connectors	Usable with proper design[m]	Harness
Signal conditioning and control unit	Unique and necessary[n] to MMS vehicle	

will be required in addition to the hardware defined in Table A-2. (This hardware is defined in paragraph 4.2.)

Figures A-1 and A-2 illustrates the C&DH Subsystem module for use with the NASA STDN and AFSCF ground networks, respectively.

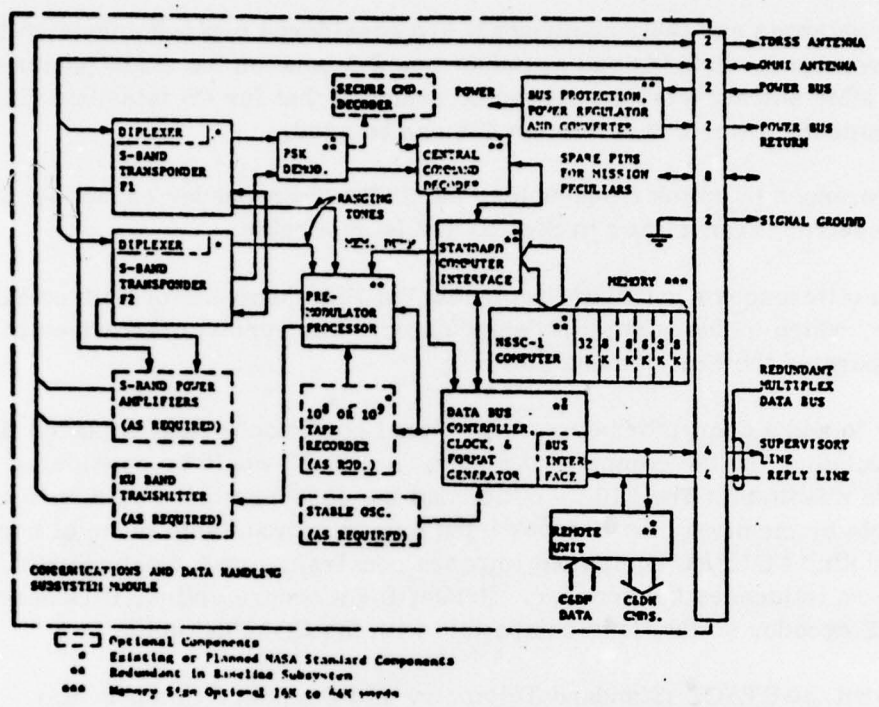


Figure A-1. C&DH Subsystem For Use With NASA STDN

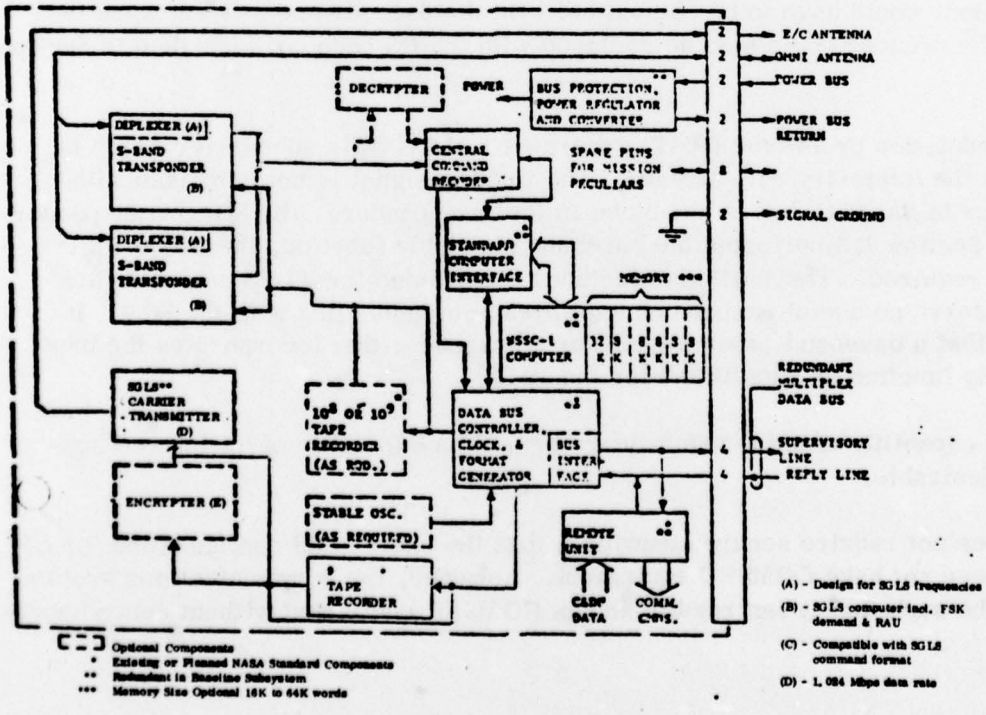


Figure A-2. C&DH Subsystem For Use With AFSCF

[a]The antenna or antennas and their components are considered payload unique on the MMS. The requirement for hybrids and switches would depend on the exact placement and design of the MMS antenna system. It can be assumed that for CS missions that the CS Turnstile antenna described in Section 4.2 can be used.

[b]Because of differences in uplink frequencies, the STDN Transponder cannot be used for SGLS. The receiver/Transmitter in Section 4.2 is applicable.

[c]The modulation differences necessitate replacing the PSK demodulator with an SGLS signal conditioner, which includes an FSK demodulator. The receiver described in Section 4.2 incorporates the FSK demodulator.

[d]It is necessary to add a decrypter between the signal conditioner (that replaces the MMS PSK de-modulator) and the command decoder. A KIR 23 would be considered appropriate for CS missions. The KIR 23 output and the decoder input would have to be made compatible by modifying the decoder input or adding a suitable piece of hardware. Further, uplink COMSEC equipment imposes constraints on the command sord format which in turn influences the decoder. Hence, for a secure uplink, it is necessary to modify the MMS decoder so that it is compatible with the COMSEC unit.

[e]These items form the STACC (Standard Telemetry and Command Components) central unit.

[f]The MMS command decoder can be retained for clear uplinks. However, the AFSCF command format would have to be compatible with the decoder and new software is required. The decoder could also be replaced with the CS decoder described in Section 4.2.

[g]The premodulation processor (PMP) generates a 1.024 MHz subcarrier which is modulated by the telemetry data stream. The ranging signal is not combined with the subcarrier in the PMP but is combined in the transponder. The SGLS transponder described in Section 4.2 performs the baseband assembly function, therefore PMP would not be required. The PMP also includes electronics for TDRS compatibility which would serve no useful purpose on satellites communicating with the SCF. It is desirable that a baseband assembly unit or transponder that incorporates the baseband assembly function be substituted for the PMP.

SGLS has a capability of using two subcarriers. The capability of having two subcarriers is desirable.

[h]The CS does not require secured downlink; thus the basic MMS configuration for CS application need not have COMSEC equipment. However, the communications system design must be such that it can readily accept COMSEC equipment without costly modifications.

For those missions requiring secured downlink, COMSEC equipment must be added to the MMS between the telemetry format generator and the pre-modulation processor for downlink protection. The spacecraft must comply with Tempest requirements to protect the classified data. Proper design practice will provide a high degree of confidence that Tempest requirements can be satisfied with little or no modification. There should be 90 dB isolation between the data and the clock; the input signal leads should be well shielded; the input and output signal leads should be run in separate cables and connectors. The encryption unit would be GFE.

[i] The MMS tape recorder has sufficient capacity and should satisfy the CS requirements.

[j] The MMS telemetry format and data rates offer a great deal of flexibility and can be used by CS. However, there may be some penalties involved in living with the fixed minor frame length (128 word), the fixed number of sub-commutated words (4), and the fixed major frame length (128 minor frames). Supercommutation of the minor frame words and/or of the subcommutated data is provided in the MMS design and will add to the flexibility.

[k] The MMS computer handles attitude control as well as C&DH. Therefore adequate room is available in the MMS computer for CS data handling.

[l] The MMS remote unit is usable for CS assuming that the data bus controller, clock and format generator, and standard computer interface used is that of the MMS. Using a CS bus controller rather than these units would require using a CS data interface unit.

[m] Assumes an initially compatible design.

[n] Involved with solar panel deployment on MMS and is required. The CS vehicle has nothing comparable. It can be assumed that the changes which must be made in the decoder will not jeopardize this function.

[p] These items are functionally described by the telemetry unit in paragraph 4.2.

APPENDIX B

EXPERIMENT THERMAL CONTROL DESCRIPTION

APPENDIX B

EXPERIMENT THERMAL CONTROL DESCRIPTION

Thermal control concepts for all experiments and the EPM were established based on the power duty cycles and configurations defined in Section 4.3. The experiment thermal control was, in general, independent of the STR and its components. However, heater power requirements defined were used to calculate the total STR power requirements and the STR battery module dissipation was established assuming 25% of the total electrical watts required would be dissipated in the battery module. Therefore the STR heat and energy rejection requirements defined included experiment requirements.

Experiment Pointing Mount Assembly (EPM)

The EPM is used with experiments 2, 5, 8 and 9. The EPM data is as follows:

- Gimbal Assembly Weight = 450 pounds
- Torque Motors
  - Roll (2) weigh 125 pounds and dissipate 224 watts (112 watts each) when operating
  - Pitch (2) weight 37 pounds and dissipate 70 watts (35 watts each) when operating
- Electronics - weighs 23 pounds and dissipates 180 watts operating and 5 watts standby
- Gyro - dissipates 35 watts continuously

In total the EPM dissipates 509 watts operating and 40 watts in standby. For 1% operation, the average power dissipation is 44.7 watts and the energy expended in 7 days is 7.51 kWh. The thermal design concept utilizes a passive design which dissipates the average power dissipation using a radiator with a 5 mil Teflon over silver thermal coating and the thermal time constant of the EPM to limit temperature uses during maximum 1 minute periods of operation. With an effective 40% view to space and incident sunlight blocked by the experiment being supported, a minimum

temperature of 10°C will be maintained at the minimum 40 watt heat dissipation load with a 3.78 square foot radiator area. At the maximum average heat dissipation of 44.7 watts, an average temperature of 18°C results. The thermal capacitance of the assembly will keep the maximum temperature during 1 minute operation periods below 30°C.

Experiment 1 (Lasercom Receiver Test)

The experiment is completely enclosed in multilayer insulation, except for the opening if there is no door. During non-operation periods, the standby power is 100 watts. With no door 204.7 watts are required during non-operation (space facing) to maintain 10°C. With a door 99.7 watts are required. Therefore, with a door the 100 watts standby power is sufficient, while with no door, an additional 104.7 watts of heater power is required. The experiment is never allowed to get sunlight directly into the optics. The duty cycle can be 30 minutes per orbit for two consecutive orbits. With an initial temperature of 10°C, the experiment will heat to 13.8°C with 30 minutes of operation at 1000 watts. During the 60 minutes prior to the next 30 minute operation, the experiment will cool to 12.1°C and then heat back up to 15.8°C. At the end of the 30 minute operation period, the experiment will cool to 10°C in 5.4 hours, at which time 104.7 watts of heater power (in addition to 100 watts standby) is required to maintain 10°C until the next operational cycle. Therefore 12.1 kWh of heater power is required with no door.

Experiment 2 (HIRISE)

It was assumed that 50% of the experiment power during operation (210 watts) is dissipated by the storage module and data processing box which are mounted on the STR, and temperature controlled by the STR heat distribution and rejection system. The remainder of the experiment power dissipation during operation (210 watts) and the standby power of 10 watts is dissipated in the optics. The optics assembly is completely enclosed in multi-layer insulation except the optics operature if there is no door. During space facing non-operation, 1089.4 watts is required to maintain 10°C with no door and 144.0 watts is required with a door. The corresponding heater power required is

1079.4 watts and 134.0 watts respectively. During operation at the orbit subsolar point for 9 minutes (10% duty cycle or 45 watt-hours per orbit), the assembly heats to 10.8°C. Subsequent to operation, the assembly will cool to 10°C in .108 hours if space facing with no door, or .815 hours with a door. The heater energy required would be 150.0 kWh with no door and 8.0 kWh with a door.

Experiment 3 (ATLAS)

The experiment is radis pointing during nightime (only) operation every orbit. Nightime is 36.4 minutes of a 90 minute orbit. Evaluations indicate that the experiment can not face the sunlight during non-operating periods without a door that is closed. The electronics units (transmitter and power supply) are mounted to and thermally isolated from the STR bridge. It is assumed that they each dissipate 50% of the experiment electronic dissipation during operation (50 watts each). They are enclosed in multi-layer insulation except for the top of the unit which has a heat rejection window utilizing a 5 mil teflon over Silver thermal coating. The transmitter has a 1.5 square foot heat rejection area. For a maximum environment orbit, the transmitter will cycle from 17.7°C. to 25.5°C. With a space facing environment during the non-operating period the unit would cool to 7.2°C. It would reach 10°C in 0.76 hours and require 39.7 watts of heater power to maintain 10°C prior to the next operation. This would require 0.6 kWh of electrical energy. The power supply, which is heavier and has a larger thermal time constant, requires 1.5 square feet of radiation area and will vary in temperature from 17.7°C to 19.1°C in a maximum environment. If space facing during the non-operating period, it will cool to 16.1°C and therefore requires no heater power. The receiver is completely enclosed in multi-layer insulation except for the aperture and a circumferential heat rejection area around the periphery. This strip is 2 feet in diameter by 12.1 inches long with no door and 19.8 inches long with a door based on maintaining a maximum average orbit temperature of 25°C. An operational power dissipation duty cycle of 300 watts for 18.2 minutes followed by 600 watts for 18.2 minutes following by non-operation for 53.6 minutes was evaluated. In a maximum orbit environment the receiver will vary from 22.7°C initially to 23.8°C after 18.2 minutes at 300 watts to 30°C after 18.2 minutes at 600 watts to 22.7°C after 53.6 minutes non-operational. If space facing during the 53.6 minute cooldown, the assembly will cool to 10°C in

0.85 hours and require 289.3 watts for .04 hours per orbit to maintain 10°C, or 1.3 kWh. The reduced power requirements with a door were not evaluated since this experiment did not represent a limiting power requirement even with no door.

Experiment 4 (Precision Release Accuracy Test)

The experiment consists of 10 cylindrical or conically shaped objects which are mounted to and thermally insulated from the STR bridge. Each is enclosed in multilayer insulation and requires 10 watts of heater power to maintain 10°C while facing space. For equally spaced operations, one operation of 5 minutes will occur every 16.8 hours. If the objects are initially at 20°C, they will cool to 10°C in 4.9 hours and each object will require 10 watts of heater power thereafter until release at 16.8 hours intervals. The heater energy required is 8.8 kWh.

Experiment 5 (ELMS)

The experiment, with an experiment pecular cryogen supply, operates continuously with a continuous nadir scan and has a constant heat dissipation. The electronic components are conductively coupled to the STR bridge and individually insulated with multi-layer insulation. The insulation on the STR bridge has openings for heat rejection near the boxes and a 5 mil Teflon over silver heat rejection coating is utilized. No heater power is required. The telescope has 9 watts internal dissipation plug the operature heat load. A circumferential radiator will be used around the telescope periphery to maintain on orbit average temperature of 20°C. The telescope thermal time constant will limit orbital temperature variations to about $\pm 5^{\circ}$C. No heater power is required.

Experiment 6 (LASS II)

The experiment has an A and B section. The B assembly is thermally remote and is self sufficient with independent power supply and thermal control capability. Both A & B experiments are on continuously during orbit nightime with continuous nadir pointing. The A components are thermally grouped together, thermally coupled to the STR bridge, with temperature maintained via one heat rejection area with a 5 mil Teflon over silver thermal coating.

Experiment 7 (XUV)

The experiment consists of one Auroral Monitor, 2 Sky monitors and a Zenith sky monitor. Temperature control from 10°C to 20°C is preferred with -20°C to 35°C acceptable. The Auroral Monitor dissipates 5 watts continuously, is box shaped, and is deployed and thermally remote from the vehicle. The box will be completely enclosed in multi-layer insulation except for a small heat rejection window, with a 5 mil Teflon over Silver thermal coating, on each of the 6 sides. In a cold environment with no external flux and 5 watts dissipation, an average temperature of 10°C is maintained with a 2.4 square inch heat rejection opening on each side. In a maximum environment with the sun normal to one side and earth plus albedo incident on the other sides, the maximum average temperature is 23.5°C. No heater power is required. The sky monitors (each 7 watts) are isolated thermally from the STR bridge and enclosed in multilayer insulation. A heat rejection window with a 5 mil Teflon over silver thermal coating is located on the top of the unit. In a cold environment, 10°C is maintained with a 22.5 square inch heat rejection area. In a maximum environment with normal sun the maximum temperature is 32.2°C, thermal capacitance will reduce this temperature range which can be adjusted between -20°C and 35°C as desired. No heater power is required. The thermal approach for the Zenith sky monitor (5 watts) will be similar to the sky monitor, with a smaller heat rejection area requirement.

Experiment 8 (OGAO)

The experiment is nadir oriented with 1 hour periods of operation (10% total duty cycle) in areas of interest. The experiment is completely enclosed in multilayer insulation except for the aperture and a 1.73 inch wide heat rejection strip around the rectangular perimeter. The heat rejection coating is 5 mil Teflon over silver and an average temperature of 25°C results. In a maximum environment the assembly temperature varies from 22.2°C initially to 27.2°C after 1 hour of operation to 22.2°C at the end of a 1.5 hour orbit. In a cold environment post operation, the assembly will cool to 10°C in 1.3 hours and require 149.4 watts of heater power (19.3 kWh) to maintain 10°C. With a door that closes at 10°C, 44.4 watts of heater power (5.7 kWh) are required.

Experiment 9 (Teal Ruby)

The experiment is completely enclosed in multi-layer insulation and isolated by conduction from the EPM. The outer experiment wall (under insulation) temperature required is 290 ±5°K. The experiment operation periods at 312 watts average dissipation are 20 minutes each with a maximum of 40 minutes operation per orbit, 40 minutes operation per day and 80 minutes operation per 7 days. A 50 watt warm up heater is activated for 19 hours prior to operation and a minimum 15 watt heater during other non-operational periods was initially identified. Four equal 42 hour operational cycles were assumed. For the estimated external insulation area of 82.9 square feet, 50 watts at steady state will result in an external wall temperature at the maximum temperature allowable of 295°K. For 50 watts to heat the experiment to 285°K (the minimum external wall temperature in 19 hours, the initial temperature when the 50 watts is activated must be a minimum of 281.8°K, with 41.5 watts required to maintain 281.8°K in a cold environment. With an initial temperature of 294.1°K (21.1°C) at launch and a cold environment the wall will cool to 283°K in the 22.7 hour period prior to the 50 watt heater operation, will heat to 285.2°K with heater operation at 50 watts for 19 hours, and will heat to 286.5°K after a 20 minute operating period. In a cold environment the wall will cool to 281.8°K from 286.5°K in 7.25 hours and will require 41.5 watts to maintain 281.8°K until the next warm up period. (This 41.5 watt requirement and the subsequent energy is greater than the 15 watt heater specified.) A special case was considered in which the experiment operated only once for 20 minutes at the end of a 7 day mission with a prior 19 hour warm up period. The 41.5 watt heater was required 16.7 hours into the mission (time to cool from 294.1°K to 281.8°K). However, the total power and phase change energy required for this case (16.0 kWh and 6.7 kWh) was less than that determined (18.8 kWh and 9.8 kWh) for the 42 hour cycle previously evaluated, so a mission duty cycle with maximum allowable payload operation is the limiting case.

Experiment 11 (Optical Countermeasures Demonstration)

The deployable component of the experiment requires no thermal control (but could use the same concept as the Auroral Monitor of Experiment 7 if required). It was assumed

that each of the 3 radiometers dissipates 1/3 of the 50 watts experiment power (16.7 watts each) during 5 minute operation periods. The radiometers are thermally isolated from the STR bridge and completely enclosed in multi-layer insulation except for the aperature. During non-operating periods with a cold environment each radiometer requires 2.4 watts of heater power to maintain 10°C. During 5 minute operating periods the radiometer will heat to 11.5°C and cool to 10°C in .43 hours subsequent to operation.

Experiment 12 (Deployable Spacecraft)

The spacecraft has its own power supply and thermal control and was considered independent (thermal and power) from the STR for this analysis.

APPENDIX C

DETAIL STRUCTURAL DESIGN REQUIREMENTS AND STRESS ANALYSIS

APPENDIX C

DETAIL STRUCTURAL DESIGN REQUIREMENTS AND STRESS ANALYSIS

The design conditions corresponding to all like phases include all factors that influence the structural design. These typically include the acceleration, thermal and dynamic environments summarized in Table C-1. The effects of structural misalignment, dimensional tolerances, thermal stress, fatigue, and creep deformation are considered in the analysis where applicable. The design loads are obtained by applying factors of safety to the limit loads. Limit loads are the two sigma flight loads. The design load factors of safety shown in Table C-2 are applied to the loads to obtain the structural design yield and design ultimate loads.

Design Conditions and Environment

The launch environments are considered most significant to the STR structural design. All loads, or load factors are given as limit loads unless otherwise designated.

Structural Load Requirements During Shuttle Flight

The primary structure has been designed to quasi-steady state accelerations. The final design loads will be determined from coupled STS/STR dynamic response analysis and dynamic analyses of other critical conditions.

Quasi-Steady Load Requirements

Quasi-steady acceleration levels have been used to size the STR structure. These preliminary load factors are based on JSC07700 Vol. XIV "Shuttle Payload Accommodations." Tables C-3 through C-5 have been reproduced from Section 7 of the aforementioned document for reference. A review of these tables reveals that the most critical condition is the emergency landing case (Table C-5). Thus, the preliminary levels to be used are shown in Table C-6.

Load factors for design of component mounting are summarized in Table C-7. These factors reflect spacecraft dynamic responses and experience gained from dynamic response analyses and dynamic test results of similar structures.

Table C-1. Mechanical Environments for Structural Design

Ground and Prelaunch Environments
Dynamic quasi-static and static loads
Vibration due to air, truck transportation and handling
Shock due to drops during handling
Launch and Ascent Environments
Steady acceleration due to thrust and maneuver forces
Transient loads due to shuttle stage startups and shutdowns, gusts and other aerodynamic excitations
Vibration and acoustic noise at liftoff, and during maximum dynamic pressure
Test Environments
Shock, vibration, acoustics, and static loads
Re-Entry, Landing, and Crashing Environments
Steady accelerations and transients due to re-entry landing, braking, and crash
Shock due to re-entry landing and crash

Table C-2. Design Load Factors of Safety

Load Condition	Design Load Factors of Safety	
	Yield	Ultimate
Launch	1.00	1.25
Transportation Handling Hazardous Conditions	1.25	2.00
Not Hazardous to Personnel	1.25	1.50
Spacecraft Structure Critical to Alignment	1.15	1.25

Table C-3. Cargo Limit Design Load Factors for 65 KLB (29484 Kg) Up and 32 KLB (14515 Kg) Down

CONDITION	LINEAR - g			ANGULAR - RAD/SEC²		
	X	Y	Z	X-X	Y-Y	Z-Z
Lift-off	-0.1 -2.9	+1.0 -1.0	+1.5 -1.5	+0.10 -0.10	+0.15 -0.15	+0.15 -0.15
High-Q Boost	-1.6 -2.0	+0.5 -0.5	+0.6 -0.6	+0.10 -0.10	+0.15 -0.15	+0.15 -0.15
Boost-Max. LF (Stack)	-2.7 -3.3	+0.2 -0.2	-0.3 -0.3	+0.20 -0.20	+0.25 -0.25	+0.25 -0.25
Boost-Max. LF (Orb Alone)	-2.7 -3.3	+0.2 -0.2	-0.75 -0.75	+0.20 -0.20	+0.25 -0.25	+0.25 -0.25
Entry and Desc Pitch Up	+1.06 -0.02	0 0	+2.5 -1.0	+0.25 -0.25	+0.75 -0.75	+0.30 -0.30
Entry and Desc Yaw	+0.75 +0.75	+1.25 -1.25	+1.0 +1.0	+0.25 -0.25	+0.30 -0.30	+0.75 -0.75
Landing	+1.0 -0.8	+0.5 -0.5	+2.8 +2.2	+0.25 -0.25	+1.25 -0.75	+0.30 -0.30

Sign convention follows that of the Orbiter coordinates system. Angular accelerations follow the right hand fule.

Ascent and landing conditions include dynamic transients effects but do not include the dynamic response of the payload.

For payload weights between 32 KLB (14515 Kg) and 65 KLB (29484 Kg), use a linear interpolation between the load factors given in Tables 7.11 and 7.12 for the entry and landing.

Payload designers should take advantage of the baseline normal flight design criteria.

Table C-4. Cargo Limit Design Load Factors for 65 KLG (29484 Kg) Down in an Emergency Situation

CONDITION	LINEAR - g			ANGULAR - RAD/SEC²		
	X	Y	Z	X-X	Y-Y	Z-Z
Entry and Desc Pitch.	+0.92 -0.02	0 0	+2.18 -0.87	+0.22 -0.22	+0.65 -0.65	+0.26 -0.26
Entry and Desc Yaw	+0.65 +0.65	+1.09 -1.09	+1.0 +1.0	+0.22 -0.22	+0.26 -0.26	+0.65 -0.65
Landing	+0.87 -0.70	+0.44 -0.44	+2.35 +1.7	+0.22 -0.22	+1.09 -0.65	+0.26 -0.26

Sign convention follows that of the Orbiter coordinate system. Angular accelerations follow the right hand rule.

Ascent and landing conditions include dynamic transients effects but do not include the dynamic response of the payload.

For payload weights between 32 KLB and 65 KLB, use a linear interpolation between the load factors given in Tables 7. 11 and 7. 12 for the entry and landing.

Table C-5. Emergency Landing Design Load Factors

CONDITION	Load Factor 32 KLB (14515 Kg) Down			Load Factor 65 KLB (29404 Kg) Down		
	X	Y	Z	X	Y	Z
Emergency Landing	+4.5 -1.5	+1.50 -1.50	+4.5 -2.0	+4.50 -0.738	+0.738 -0.738	+2.215 -0.985
Emergency Landing (Crew Compartment Interior)	+20.0 -3.3	+3.3 -3.3	+10.0 -4.4			

Sign convention follows that of the Orbiter coordinate system.

Emergency landing load factors are ultimate. The longitudinal load factors are directed in all aftward azimuths within a cone of 20 degrees half-angle. The specified load factors shall operate separately.

For payload weight between 32 KLB 914515 Kg) and 65 KLB (29484 Kg), use a linear interpolation between the load factors given.

Table C-6. STR Quasi-Steady Loop Factors

Condition	X (g)	Y (g)	Z (g)
Emergency Landing	+4.5	-	-
	-1.5	-	-
	-	±1.5	-
	-	-	+4.5
	-	-	-2.0

These are ultimate load factors and act separately.

Table C-7. Limit Load Factor for Component Mounting

Component Zone	Longitudinal (g)	Lateral (g)
Equipment Modules	35	30
Propellant Tanks	30	20

Acoustic and Vibration Levels

Acoustic and vibration levels have been established for both system and components to be used during tests and dynamic analyses to determine response accelerations. Table C-8 presents the acoustic requirements. Random vibration levels are not specified, since acoustic testing will be performed for the STR. Acoustic testing better represents the booster environment and therefore results in a more realistic test program. The acoustic levels apply for both the spacecraft and components.

All STR components will be qualified, at the component level, to vibration and/or acoustic test levels. Component random vibration levels are presented in Table C-9. These levels are specified, per the component zone location. Dynamic environment component zones are defined in Table C-10. The random component vibration requirements for the Shuttle acoustic levels environments were derived using the results of a current GE cost effectiveness study. The method of derivation uses the statistical analysis of measurements from six spacecraft and a random response spectra approach to assure the adequacy of spectrum smoothing. The derivation follows the basic MIL-STD-1540A procedures.

It should be noted that extensive studies are underway by various organizations to better define the Space Shuttle environment. The acoustic levels presented in Table C-8, are based on a shuttle bay design goal of 145 dB overall (151 dB for qualification levels). This level was not based on a rigorous analysis since the shuttle structural design and the launch pad final configuration had not been firmed-up at the time the goal was established. Further analyses by RI, JSC and BB&N have indicated that the acoustic levels may be significantly lower in the high frequency range. Because the

Table C-8. Acoustic Noise (Qualification Level)

Octave Band Center Frequency (Hz)	Sound Pressure Level: dB Ref. 20 N/M^2
31.5	133
63	139
125	143
250	145
500	145
1000	143
2000	140
4000	136
8000	132
Overall	151
Duration (Seconds)	120

internal acoustic environment and acoustic properties of the payload bay structure are not completely defined at this time, the acoustic environment definition requires continual attention.

Rigidity Requirements

Rigidity requirements have been established for both the STR structure and its components. Minimum resonant frequency requirements have been imposed to limit dynamic load amplifications. Shuttle payload studies have shown that payloads can exhibit strong dynamic coupling with the shuttle bending mode near 5 Hertz, a wing bending mode near 10 Hertz and various Shuttle modes in the 13 to 18 Hertz range. Consequently, the minimum resonant frequencies defined in Table C-11 have been established for the STR design and particular frequency ranges to be avoided have been indicated.

Table C-9. Component Random Vibration Levels

Zone	Frequency Range (Hz)	Power Spectral Density (g^2/Hz)
1 & 2	20 - 500	3 dB/Oct
	500 - 1000	.25 g^2/Hz
	1000 - 2000	-6 dB/Oct
	Overall	17.67 g rms
3	20 - 100	+6 dB/Oct
	100 - 500	.5 g^2/Hz
	500 - 2000	-3 dB/Oct
	Overall	23.7 g rms
4	20 - 100	+6 dB/Oct
	100 - 1000	.5 g^2/Hz
	1000 - 2000	-6 dB/Oct
	Overall	26.8 g rms
Test Duration 4 Min/Axis		

Note: The vibration shall be applied along three mutually perpendicular directions.

Table C-10. Definition of Dynamic Environment Zones

Zone	Definition
1	Primary structure within two feet of the STR/STS interface.
2	Primary structure more than two feet from the STS interface.
3	Secondary structure internal to the outer shell.
4	Secondary external structural.

Table C-11. Minimum Resonant Frequencies

Mode	Frequency (Hz)
Fundamental*	6.5 Hz
Secondary Structure	100 Hz
*Cantilevered from orbiter attachments with payload attached. Major resonances should also avoid the 13 to 18 Hz frequency range.	

The secondary structure component mounting resonant frequency requirement is included so that a generally stiff structure with predictable loads can be obtained. In addition, it precludes possible clearance problems between components and the spacecraft structure. For most of the components, a minimum of 100 Hertz component mounting resonant frequency has been set to ensure adequate separation between the component resonances and the fundamental spacecraft modes. This allows design and testing of the components and mounting brackets to vibration levels that are essentially independent of the fundamental structural resonances.

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STANDARD TEST RACK

THE TEST RACK IS MADE FROM 6061-T6 ALUMINUM. THE MATERIAL PROPERTIES WERE TAKEN FROM MILHDBK 5B.

THE FORCES AND STRESSES USED IN THE FOLLOWING COMPUTATIONS WERE OBTAINED FROM THE NASTRAN COMPUTER OUTPUT.

THE FOLLOWING IS A LIST OF REFERENCES USED:

1.) Sechler, Ernest E., and Lois G. Dunn: AIRPLANE STRUCTURAL ANALYSIS AND DESIGN, Dover, 1963.

2.) G.C. MARSHALL SPACE FLIGHT CENTER ASTRONAUTIC STRUCTURES MANUAL.

3.) MIL-HDBK-5B

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STANDARD TEST RACK

MATERIAL — 6061-T6 Aluminum

$F_{tu} = 42$ ksi
$F_{ty} = 35$ ksi
$F_{cy} = 35$ ksi
$F_{su} = 27$ ksi
$E = 10.5 \times 10^6$ psi

GRAPHS FOR PLATE ANALYSIS (Ref. 1)

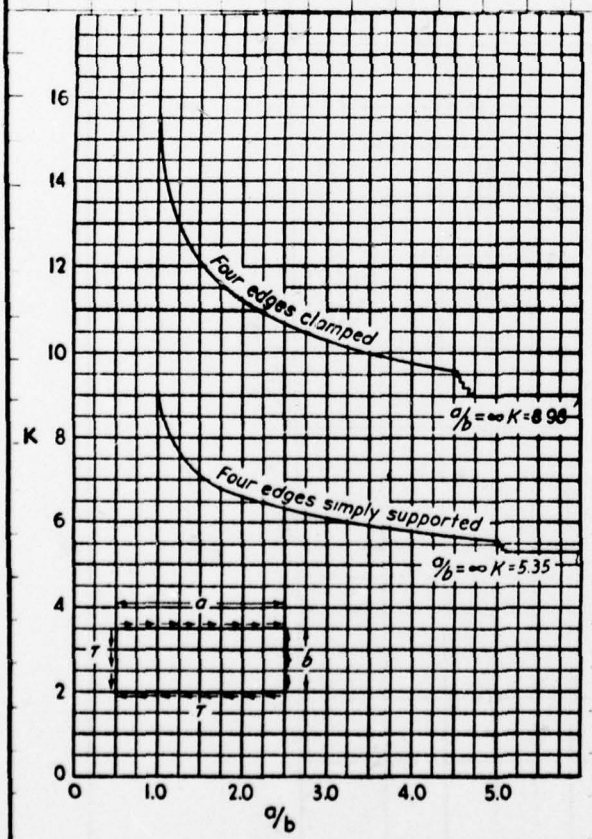


FIG. 5-11. Value of K for sheets under shear loads.

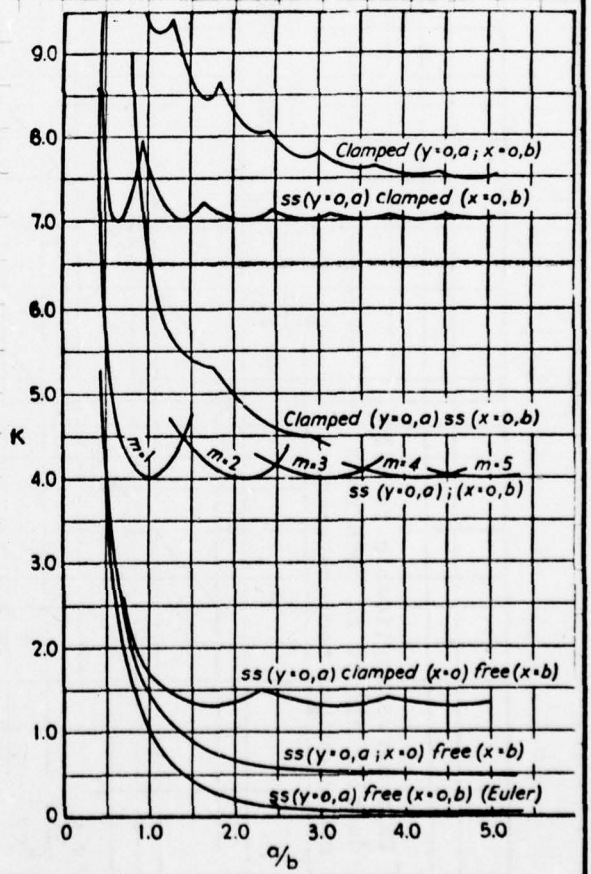
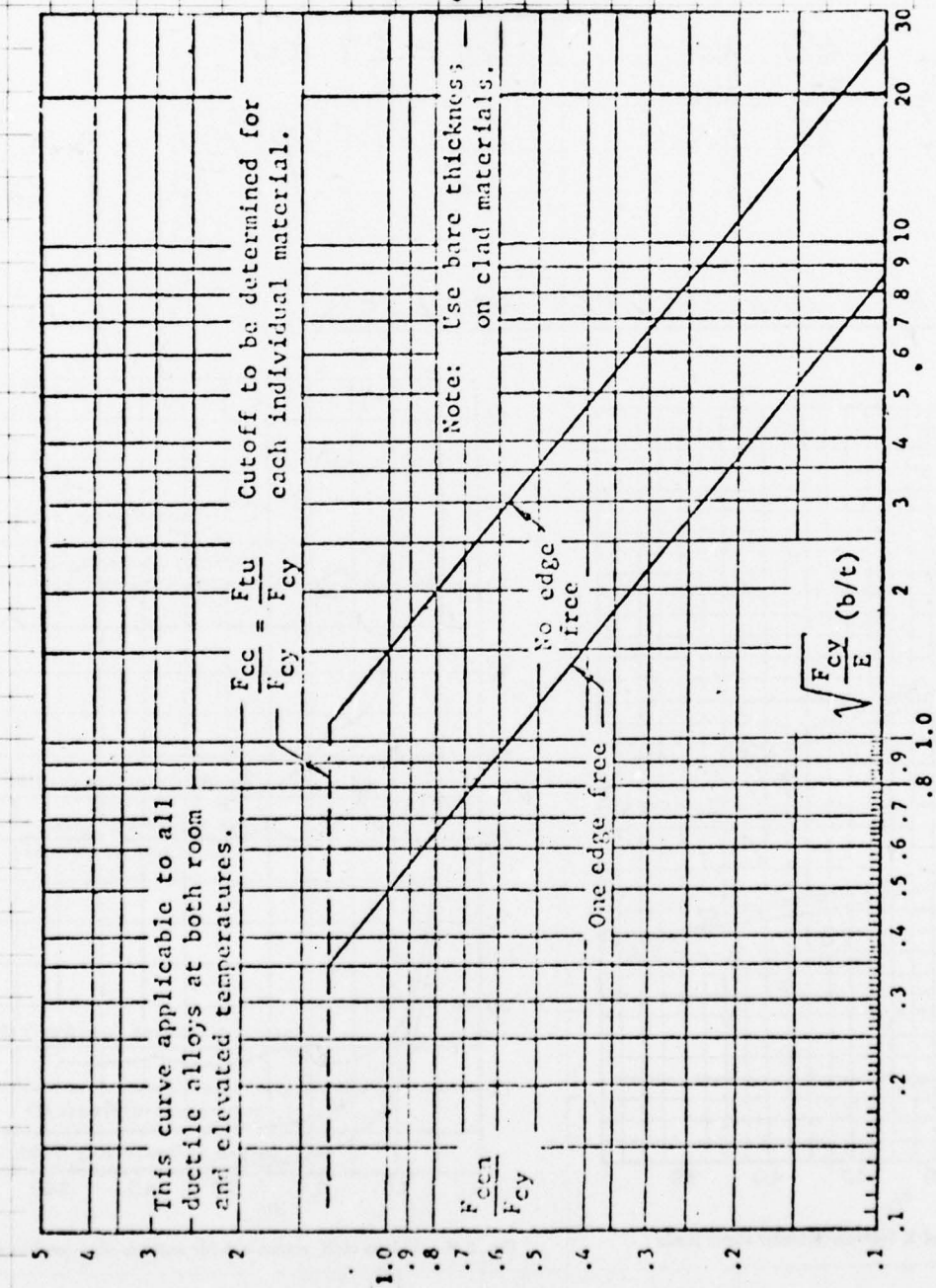


FIG. 5-8. Values of K versus a/b for various edge conditions.

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GRAPH FOR BEAM CRIPPLING ANALYSIS (Ref 2)

Fig. 10



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PLATE CALCULATIONS

THE CONSTRUCTION OF THE STR UTILIZES AS MUCH COMMONALTY AS POSSIBLE. THUS, THE PLATES USED FOR THE BRIDGE AND OUTTER SURFACE OF THE RACK ARE THE SAME. THE INNER RACK SURFACE BEING DIFFERENT.

BRIDGE AND OUTTER SURFACE:



$\frac{a}{b} = 13.11$ FROM FIGURES 5.8 & 5.11

$K_c = 4.0 \quad K_s = 5.35$

$$F_{ce} = K \frac{\pi^2 E}{12(1-\nu^2)} \left(\frac{t}{b}\right)^2$$

$F_{ce_c} = 30138.$ PSI

$F_{ce_s} = 40310.$ PSI [THIS IS GREATER THAN F_{su}. USE F_{su}]

$F_{ce_s} = F_{su} = 27000.$ PSI

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PLATE CALCULATIONS

INNER SURFACE :

$b = 5.$

$t = .1$ $\quad \frac{a}{b} = 8.356$

$a = 41.78$ $\quad K_c = 4.$ $\quad K_s = 5.35$

$\therefore$

$F_{cr_c} = 14768.$ PSI

$F_{cr_s} = 19752.$ PSI

FROM THE NASTRAN RESULTS THE MOST HIGHLY STRESSED ELEMENTS ARE

LOCATION	ELEMENT NO	LOAD CASE	σ_c PSI	τ PSI
OUTER SURFACE	9	9	5970.	2666
INNER SURFACE	2	8	2319.	1010.
BRIDGE	86	10	2764.	2588.

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PLATE CALCULATIONS

MARGIN OF SAFETY

$$M.S. = \frac{1}{R_c + R_s^2} - 1$$

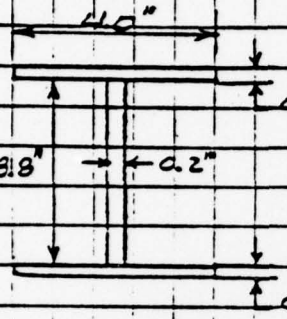
$$M.S._{OUTER} = \frac{1}{\frac{5970}{30138} + \left[\frac{2666}{27000}\right]^2} - 1 = \underline{\underline{+H16H}}$$

$$M.S._{INNER} = \frac{1}{\frac{2319}{14768} + \left[\frac{1010}{19752}\right]^2} - 1 = \underline{\underline{+H16H}}$$

$$M.S._{BRIDGE} = \frac{1}{\frac{2764}{30138} + \left[\frac{2588}{27000}\right]^2} - 1 = \underline{\underline{+H16H}}$$

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LONGITUDINAL I-BEAM



$A = 2.56\ in^2$

$I_x = 27.20\ in^4$

$I_y = 1.07\ in^4$

CRITICAL LOAD CASE 10

THE MAXIMUM STRESS OCCURS IN ELEMENT 102.
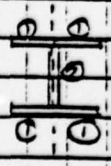
$M = 7176\ "\#$ $\quad P = 74.1\ \#$

$$\sigma = \frac{Mc}{I} + \frac{P}{A}$$

$$\sigma = \frac{7176(2.0)}{1.07} + \frac{74.1}{2.56} = 13410\ psi$$

ANALYSIS

CHECK FOR CRIPPLING (Fig. 10)



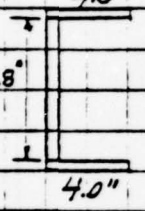
	b	t	$\frac{b}{t}\sqrt{\frac{F_{cy}}{E}}$	F_{ccn}/F_{cy}	F_{ccn}	bt	P_{ccn}
①	2	0.1	1.22	0.5	19.5 ksi	0.2	3.90 K
②	8.8	0.2	2.68	0.62	23.4 ksi	1.76	41.2 K

$$F_{cc} = \frac{\Sigma P_{ccn}}{\Sigma bt} = \frac{4 \times 3.9 + 41.2}{2.56} = 22.2\ ksi$$

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$$M.S. = \frac{F_{cc}}{\sigma} - 1. = \frac{22.2}{13.41} - 1. = \underline{1.65}$$

9 IN. DEEP CHANNEL



thicknesses = 0.1"

$A = 1.68\ in^2$

$I_x = 21.52\ in^4$

$I_y = 2.661\ in^4$

CRITICAL LOAD CASE 8

THE MAXIMUM STRESS OCCURS IN ELEMENT 94.

$M = 42618''^{\#}$ $\quad P = 247^{\#}$

$$\sigma = \frac{Mc}{I} + \frac{P}{A}$$

$$\sigma = \frac{42618(2.0)}{2.661} + \frac{247}{1.68} = \underline{32179\ psi}$$

ANALYSIS

CHECK FOR CRIPPLING (Fig. 10)



	b	t	$\frac{b}{t}\sqrt{\frac{F_{cy}}{E}}$	F_{ccn}/F_{cy}	F_{ccn}	bt	P_{ccn}
(1)	4.0	0.1	2.44	.275	10.73 ksi	.40	4.29 K
(2)	8.8	0.1	5.36	.35	13.65 ksi	.88	12.01 K

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ASSME PURE BENDING DUE TO SMALL AXIAL FORCE

USING REF 2:

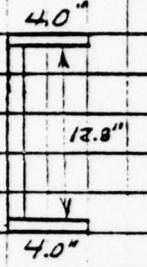
$$M_{cr} = 2\left(\underbrace{\sum F_{cc_n} b_n t_n \bar{y}_n}_{\text{For flange members}} + \underbrace{\sum F_{cc_m} b_m t_m \bar{y}_m / 2}_{\text{For web members}}\right)$$

$$M_{cr} = 2\left[4.29^{k} \times 4.45'' + 12.01^{k} \times \tfrac{2}{3}(4.4'')\right]$$

$$= 108.6''^{k}$$

$$M.S. = \frac{108.6}{42.6} - 1. = \underline{\underline{1.55}}$$

TAPERED CHANNEL



thicknesses = 0.01"

$A = 2.08\ in^2$
$I_x = 50.76\ in^4$
$I_y = 2.94\ in^4$

CRITICAL LOAD CASE 10

THE MAXIMUM STRESS OCCURS IN ELEMENT 23.

$M = 9972\ "\#$ $\quad P = 1071\ \#$

$$f = \frac{Mc}{I} + \frac{P}{A}$$

$$= \frac{9972(3.2)}{2.94} + \frac{1071}{2.08} = \underline{11369\ psi}$$

ANALYSIS

CHECK FOR CRIPPLING (Fig. 10)



	b	t	$\frac{b}{t}\sqrt{\frac{F_{cy}}{E}}$	F_{ccn}/F_{cy}	F_{ccn}	bt	P_{ccn}
(1)	4.0	0.1	2.44	.275	10.73 ksi	0.40	4.29 K
(2)	12.8	0.1	7.81	.265	10.34 ksi	1.28	13.23 K

ASSUME PURE BENDING DUE TO RELATIVELY SMALL AXIAL FORCE

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Using Ref 2:

$$M_{cr} = 2\left[4.29 \times 6.45 + 13.23 \times \tfrac{3}{3}(6.4)\right]$$

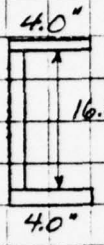
$$= 168 \text{ "K}$$

$$M.S. = \frac{168}{11.4} - 1. = \underline{+\text{HIGH}}$$

FOR COMPRESSION ONLY

$$M.S. = \frac{F_{cu}}{\sigma} - 1 = \frac{39\ ksi}{11.4\ ksi} - 1 = \underline{2.42}$$

17 IN. DEEP CHANNEL



thicknesses = 0.1"

$A = 2.48\ in^2$
$I_x = 96.64\ in^4$
$I_y = 3.13\ in^4$

CRITICAL LOAD CASE 10

THE MAXIMUM STRESS OCCURS IN ELEMENT 39

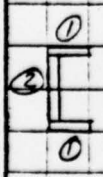
$M = 46307\ "\#$ $\quad P = 175^{\#}$

$$\sigma = \frac{Mc}{I} + \frac{P}{A}$$

$$= \frac{46307(3.321)}{3.1287} + \frac{175}{2.48} = \underline{49223\ psi}$$

ANALYSIS

CHECK FOR CRIPPLING (Fig. 10)



	b	t	$\frac{b}{t}\sqrt{\frac{F_{cy}}{E}}$	F_{ccn}/F_{cy}	F_{ccn}	bt	P_{ccn}
①	4.0	0.1	2.44	.275	10.73 ksi	0.40	4.29 K
②	16.8	0.1	10.25	.210	8.19 ksi	1.68	13.76 K

ASSUME PURE BENDING DUE TO RELATIVELY SMALL AXIAL FORCE.

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USING REF 2:

$$M_{cr} = 2(4.29 \times 8.45 + 13.76 \times \tfrac{2}{3}[8.4])$$

$$= 227 \text{"k}$$

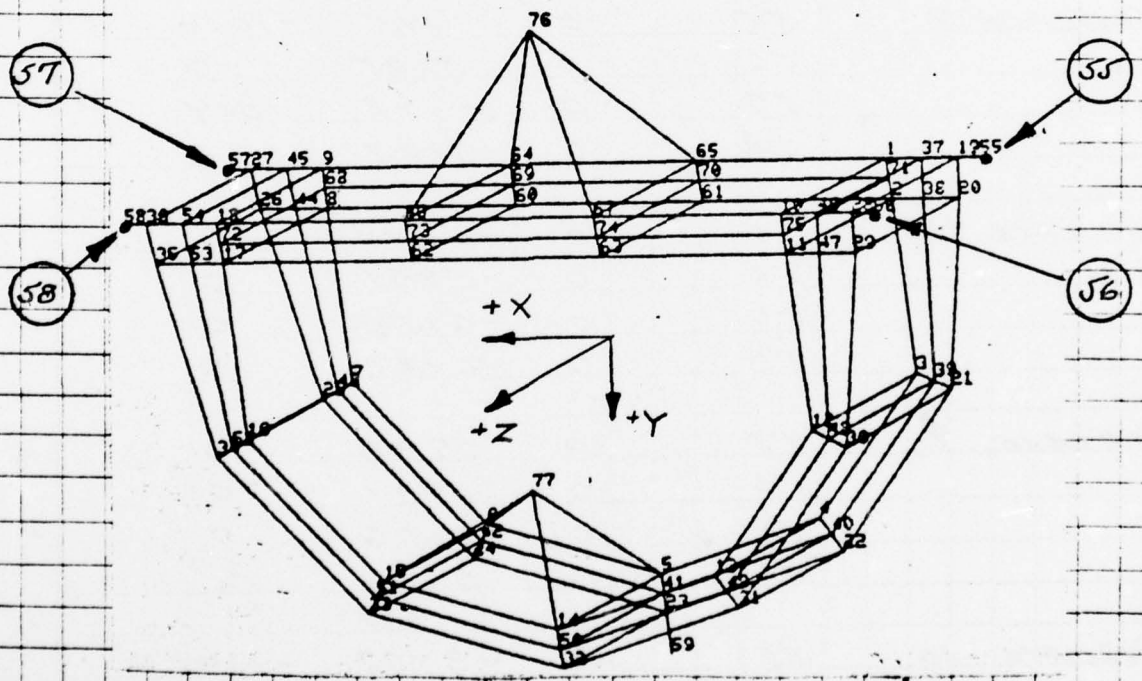
$$M.S. = \frac{227}{46.3} - 1. = \underline{\underline{3.89}}$$

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TRUNNION FITTINGS

FOUR TRUNNION FITTINGS ARE USED TO ATTACH THE STR TO THE SHUTTLE INTERFACE AND PROVIDE RESTRAINT IN THE Y AND Z DIRECTIONS.

SHUTTLE INTERFACE GRID POINTS ARE GRID POINT NUMBERS 55, 56, 57, & 58 AS SHOWN IN THE FOLLOWING SKETCH



ALL FOUR GRIDPOINTS ARE RESTRAINED IN THE Y DIRECTION

ONLY GRIDPOINTS 55 AND 57 ARE RESTRAINED IN THE Z DIRECTION

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TRUNNION FITTING

MAXIMUM LOADS (ULTIMATE) AT THE FOUR SHUTTLE INTERFACE GRIDPOINTS FOR THE DESIGN LOADING CONDITIONS ARE OBTAINED FROM THE NASTRAN FINITE-ELEMENT ANALYSIS
THE LOADS ARE SHOWN BELOW.

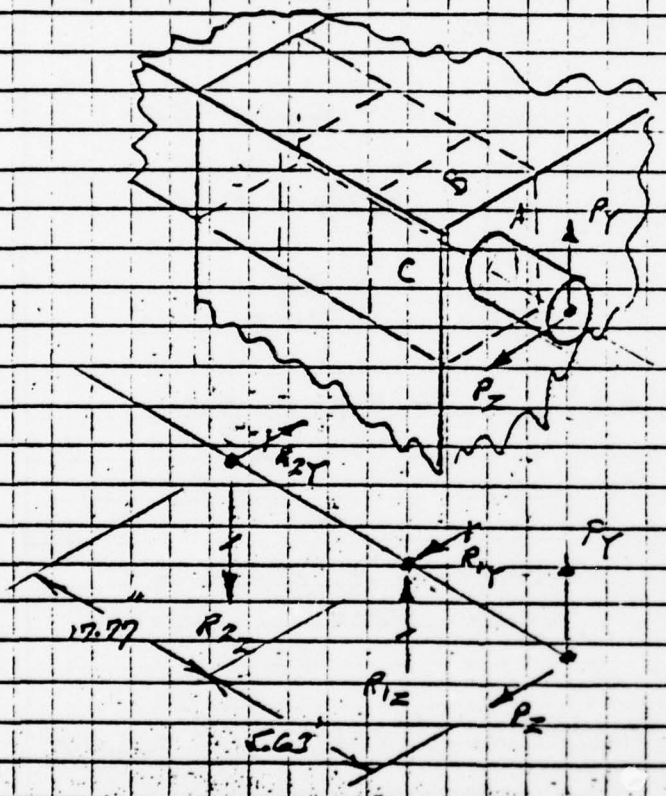
CONDITION	GRIDPOINT	LOADS (POUNDS)		
		F_x	F_y	F_z
SUBCASE 7	55	0	145	-1014
	56	0	3225	0
	57	0	-142	1014
	58	0	-3229	0
SUBCASE 8	55	0	-5998	0
	56	0	-5981	0
	57	0	-6156	0
	58	0	-6120	0
SUBCASE 9	55	0	5998	0
	56	0	5981	0
	57	0	6156	0
	58	0	6120	0
SUBCASE 10	55	0	-8433	-11979
	56	0	8483	0
	57	0	-8628	-12276
	58	0	8628	

CRITICAL LOADS OCCUR IN SUBCASE 10

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TRUNNION FITTING

THE TRUNNION FITTING LOADS ARE TRANSFERRED TO THE STR MAIN STRUCTURE BY VERTICAL FITTINGS AND SUPPORT ORDINATES AT AREAS A, B, AND C AS SHOWN BELOW.



THE TRUNNION FITTING LOADS P_Y AND P_Z ARE REACTED INTO THE STR STRUCTURE BY FITTINGS AT AREAS A, B, & C. DISTRIBUTION OF LOADS INDIVIDUALLY IS ANALYZED CONSERVATIVELY AS FOLLOWS:

P_Y LOAD — 60% TO SURFACE A } OVERLAPPING ASSUMPTION - (CONSERVATIVE)
60% TO SURFACE C }

P_Z LOAD — 60% TO SURFACE B } "
60% TO SURFACE A }

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TRUNNION FITTING

∴ FOR THE CRITICAL LOADS OF SUBCASE 10 AT GRIDPOINT 57

$P_Y = -8628$ LBS

$P_Z = -12276$ LBS

$R_{1Z} = \frac{5.63 + 17.77}{17.77}(-12276) = -16165$ LBS.

$R_{2Z} = \frac{5.63}{17.77}(-12276) = -3889$ LBS

$R_{1Y} = \frac{5.63 + 17.77}{17.77}(-8628) = -11362$ LBS

$R_{2Y} = \frac{5.63}{17.77}(-8628) = -2734$ LBS

THE TOTAL LOADS AT SURFACES A, B, AND C FOR THE R_{1Z} AND R_{1Y} LOADS SHOWN ABOVE ARE THEREFORE :

SURFACE A = .60(16165) + .60(11362) = 16,517 LBS

SURFACE B = .60(16165) = 9700 LBS

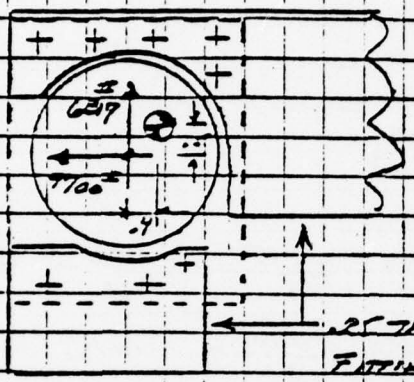
SURFACE C = .60(11362) = 6817 LBS

THE ABOVE LOADS ARE USED TO CHECK TRANSFER OF TRUNNION LOADS TO THE SUPPORTING FITTINGS. LOAD DISTRIBUTIONS ARE SHOWN IN THE FOLLOWING SKETCHES.

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TRUNNION FITTING

SURFACE A

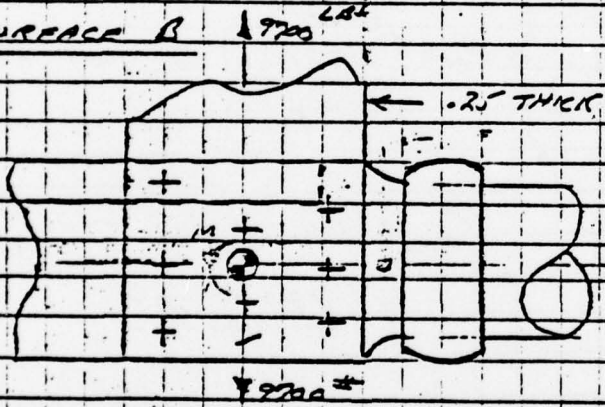


NOTE: TRUNNION OUTLINE SHOWN DOTTED.

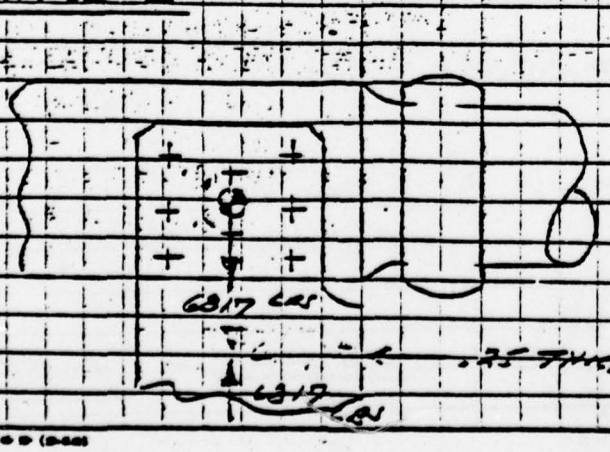
ALL LOADS REACTED BY BOLT ATTACHMENT GROUP WITH C.G. OF GROUP SHOWN AS ⊕

ALL BOLTS ARE 5/16" DIA.

SURFACE B



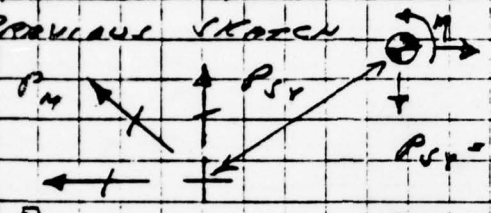
SURFACE C



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TRUNNION FITTING

MAXIMUM BOLT LOADS OCCUR IN THE SURFACE A FITTING. THE MAXIMUM LOADED BOLT IS IN THE LOWER LEFT CORNER OF THE BOLT PATTERN SHOWN IN THE PREVIOUS SKETCH



$$P_{SV} = \frac{6817}{8} = 852 \text{ LBS}$$

$$P_{SH} = \frac{9700}{8} = 1213 \text{ LBS}$$

I_p OF BOLT GROUP = 48.35 IN^2

$M_{OM} = .1(6817) + .5(9700) = 2577$ IN LBS

$$P_M = \frac{2577(3.2)}{48.35} = 501 \text{ LBS}$$

$$P_{MAJ} = 501 + \sqrt{852^2 + 1213^2}$$

$$= 1983 \text{ LBS}$$

BEARING STRAIN IN FITTING:

$$f_{br} = \frac{1983}{.25(.3125)} = 25{,}382 \text{ LBS/IN}^2$$

$F_{BR_y} = 58{,}000 \text{ LBS/IN}^2$ (REF 3 6061-T651 PLATE)

$$M.S. = \frac{58000}{25382} - 1 = +1.28$$

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TRUNNION FITTING

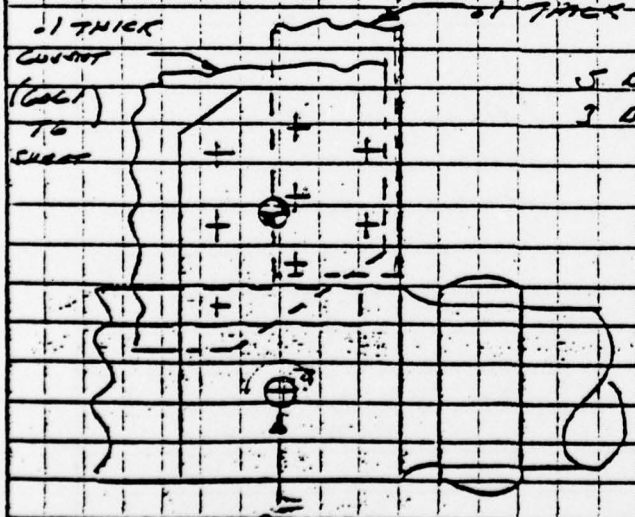
BOLT SHEAR ALLOWABLE = 6900 LBS

(A-286 MAT'L
F_{su} = 90,000 LB/IN^2)

$$M.S. = \frac{6900}{1983} - 1 = +2.48$$

BOLT LOADS IN THE FITTING (SURFACES A, B & C) ARE TRANSFERRED TO GUSSETS OR SKIN SURFACES OF THE STR MAIN STRUCTURE

CRITICAL LOADS OCCUR IN THE TRANSFER OF LOADS FROM SURFACE B



5 BOLTS IN .2 THICK MAT'L
3 BOLTS IN .1 THICK MAT'L

LOADS ARE DISTRIBUTED IN RELATION TO RELATIVE BEARING STRENGTHS

FOR BOLTS IN .1

$$P = \frac{9700}{13} = 746 \text{ LBS}$$

FOR BOLTS IN .2

$$P = 746(2) = 1492 \text{ LBS}$$

$$f_{BR} = \frac{746}{.1(.3125)} = 23,870 \text{ LBS/IN}^2$$

$$M.S. = \frac{F_{BRU}}{23870} - 1 = +1.43$$

FORM 1-0120 D (3-66)

BY JCA CK. DATE 11/17/71 REV.	GENERAL ELECTRIC	PAGE 24 MODEL REPORT

TRUNNION FITTINGS

TRUNNION BAR

THE MAX BENDING MOMENT IN THE TRUNNION BAR FITTING OCCURS FROM THE SUBCASE 10 LOADS AT GRIDPOINT 57

$M_{BM} = 5.63\left(\sqrt{12{,}276^2 + 8628^2}\right)$

$= 84{,}476$ IN LBS.

SECTION PROPERTIES

O.D. = 3.25 IN

$A = 8.2958$ IN^2

$I = 5.4765$ IN^4

$\therefore f_b = \frac{84476\,(3.25)}{2\,(5.4765)} = \underline{25066 \text{ LBS/IN}^2}$

$F_b = f_m + f_b\,(K-1)$ (ALLOWABLE BENDING STRESS BASED ON YIELD MODULUS OF RUPTURE)

$f_m = F_{ty} = 64{,}000$ LBS/IN^2 (MICHNOK 5) 7075-T6 FORGING

$K = 1.7$

$f_b = 27{,}500$ } PER BRUHN – FUNDAMENTALS OF FLT. STRUCTURES

$\therefore F_b = 83{,}250$ LBS/IN^2

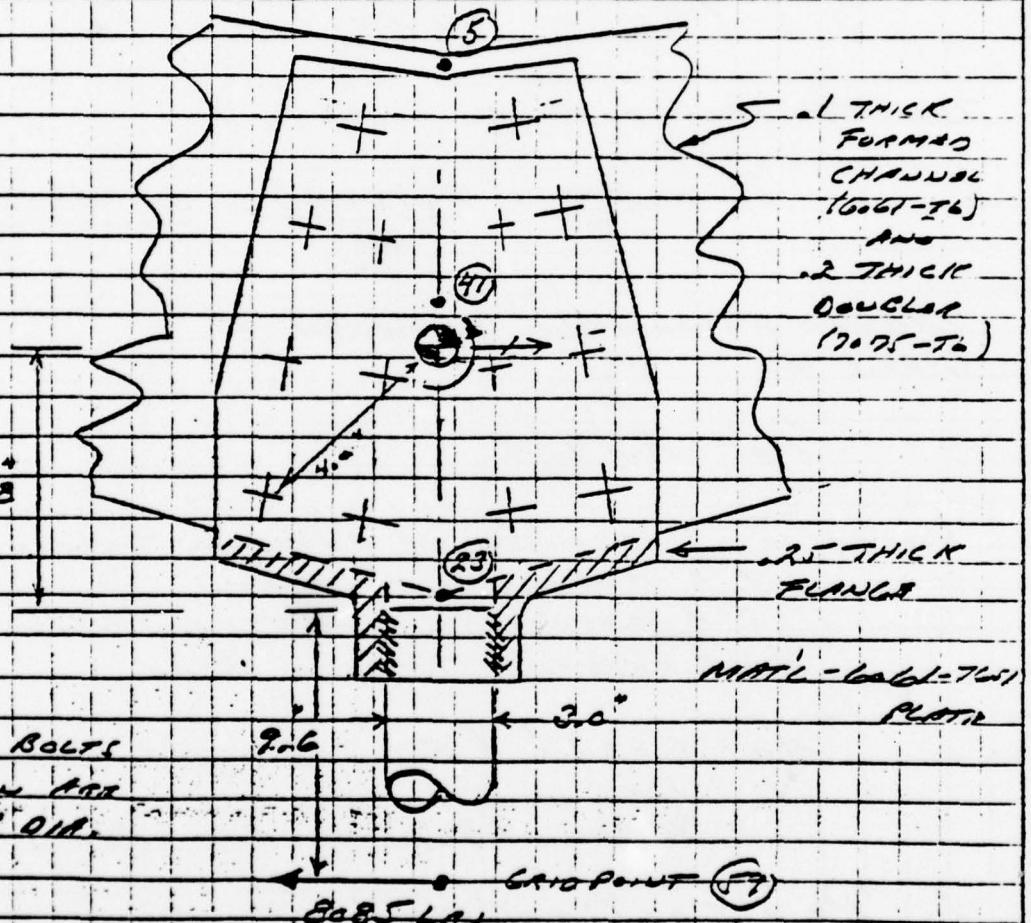
$M.S. = \frac{83250}{25066} - 1 = \underline{+2.32}$

BY OCA	GENERAL ELECTRIC	PAGE 25
CK.		MODEL
DATE 10/17/17 REV.		REPORT

KEEL TRUNNION

MAXIMUM LOADS AT THE KEEL TRUNNION FITTING INTERFACE WITH THE SPACE SHUTTLE OCCUR FOR CONDITION SUBCASE 2

P_{MAX} = 8085 LBS



THE 8085 LB LOAD IS FIRST TRANSFERRED TO THE THREAD PART OF THE FITTING. THE LOAD IS THEN TRANSFERRED THROUGH THE BOLT PATTERN INTO THE FORMED CHANNELS.

BY DCA CK. DATE 10/17/77 REV.	GENERAL ELECTRIC	PAGE 26 MODEL REPORT

KEEL TRUNNION

THREAD CHECK

THE BENDING MOMENT APPLIED TO THE THREADED AREA IS:

$$8085\,(9.6) = 77{,}616 \text{ IN LBS (MAX)}$$

THE EQUIVALENT AXIAL LOAD IS:

$$P_A = \frac{2M}{R} = \frac{2(77616)}{1.5} = 103488 \text{ LBS}$$

THREAD ALLOWABLE STRENGTH IS GIVEN BY:

$$P_E = A_e L S$$

WHERE P_E = STRENGTH OF EXTERNAL THREAD
L = ENGAGEMENT LENGTH
S = MATERIAL SHEAR STRENGTH (ULT)
A_e = EXTERNAL THREAD AREA

$A_e = 5.23 \text{ IN}^2$ (FOR 3-12 UN-2A THREAD)
$L = 1.25$ IN
$S = 46000$ (7075-T6 BAR - REF 3)

$$\therefore P_E = 5.23\,(1.25)(46000)$$

$$= 300725 \text{ LBS}$$

$$M.S. = \frac{300725}{103488} - 1 = \underline{\underline{+1.91}}$$

BY DCA	GENERAL ELECTRIC	PAGE 27
CK.		MODEL
DATE 10/17/77 REV.		REPORT

KEEL TRUNNION

BOLT ATTACHMENT CHECK

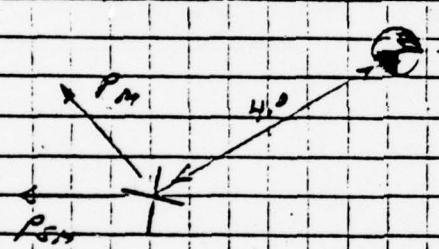
LOADS @ BOLT PATTERN C.G.

$P = 8085$ LBS

$M = 8085\,(9.6 - 4.8)$

$= 116424$ IN-LBS

MAXIMUM BOLT LOADS OCCUR IN THE LOWER CORNER ATTACHMENTS



$P_{SH} = \frac{8085}{14} = 578$ LBS

I_{POL} OF BOLT GROUP $= 125.2$ IN^2

$P_M = \frac{116424\,(4.0)}{125.2} = 3633$ LBS

$P_{RTM} \cong 3600$ LBS

BEARING STRESS IN FITTING:

$f_{br} = \frac{3680}{.25\,(.312)} = 47,105$ LBS/IN^2

$F_{BRY} = 131000$ LBS/IN^2 (REF 3 - 3.7.5-16)

$M.S. = \frac{131000}{47,104} - 1 = +1.56$

BY DCA CK. DATE 11/17/77 REV.	GENERAL ELECTRIC	PAGE 28 MODEL REPORT

KEEL TRUNNION

BEARING STRESS IN FORMED CHANNEL AND DOUBLER PLATE

THE .1 IN THICK 6061 CHANNEL AND THE .2 IN. THICK 7075 DOUBLER WILL DEVELOP A BEARING STRENGTH EQUIVALENT TO THE .25 7075 MATERIAL PREVIOUSLY CALCULATED

$\therefore$ M.S $\cong$ +1.5L

APPENDIX D

TEST MATRICES

STR Study	Environmental Test														
	Functional	Thermal Vacuum	Thermal Cycling	Sine Vibration	Random Vibration	Acceleration	Acoustic	Shock (Pyro/Trans)	Leak	EMC/ESD Mag Field	Life Test	Burn-In	Lot Sample	Humidity	Proof Burst
Communication Subsystem															
• Antennas	A				A										
• Hybrid	A														
• Diplexer	A														
• Transmitter	A	A			A										
• Receiver	A	A			A										
• Recorder	A	A			A										
• TLM Unit	D, Q, A		D, Q, A		D, Q, A										
• CMD Decoder	D, Q, A		D, Q, A		D, Q, A										
• Decryptor	(GFE)														
• Encryptor	(GFE)														
• Payload I/F Unit	D, Q, A		D, Q, A		D, Q, A										
Power Subsystem															
• Power Controller	QA		QA		QA										
• Ordnance Cont.	QA		QA		QA										
• Battery	QA		QA		QA										
- Case	-														
- Cells	QA														
• Harness	QA														
Structure Subsystem															
• STR Frame															
• Standard Panel															
• Filler Panel															
• Mechanisms															
- Sep. Nuts													A		
- Pin Pullers													A	Pyco Tests	
- Marmon Clamp	Q												A		
Thermal Control Subsystem															
• Heat Pipes	QA		QA		QA										
• Heat Exchanger	D, Q, A		D, Q, A		QA										
• Blankets	-														
• Tapes/Coatings	A														
Attitude Control Subsystem															
• Sun Sensor	A														
• Inertial Reference Unit	A	A			A										
• Star Tracker	A	A			A										
• Gimbel Assy	A														
• Drive Electronics	D, Q, A		QA		QA										
• Power Conditioner	D, Q, A		QA		QA										
• Gyro	A	A			A										
• Optical Encoder	A	A			A										
Computer	A		A												
• Power Conditioner	D, Q, A		QA		QA										

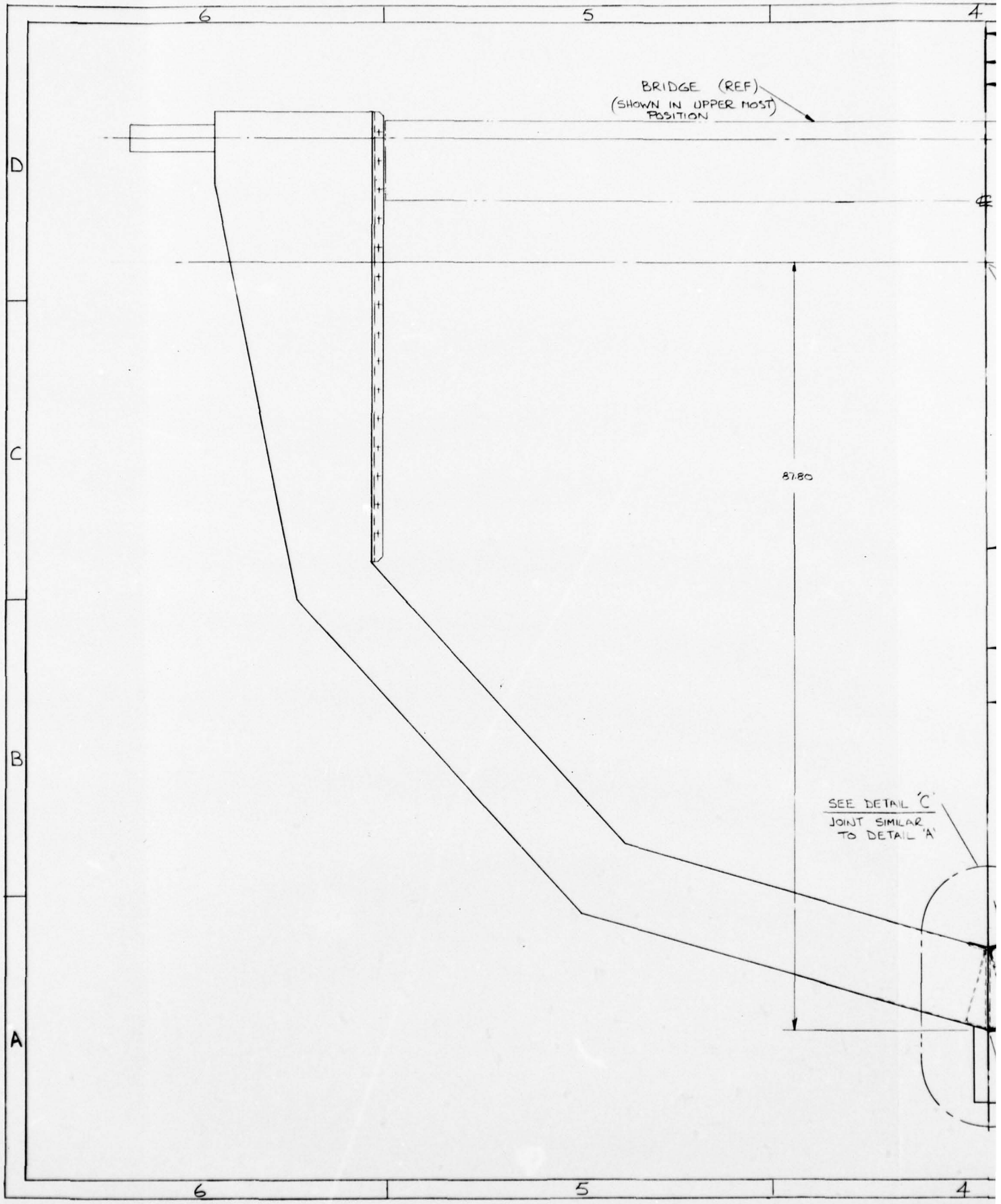
STR STUDY - EQUIPMENT HERITAGE

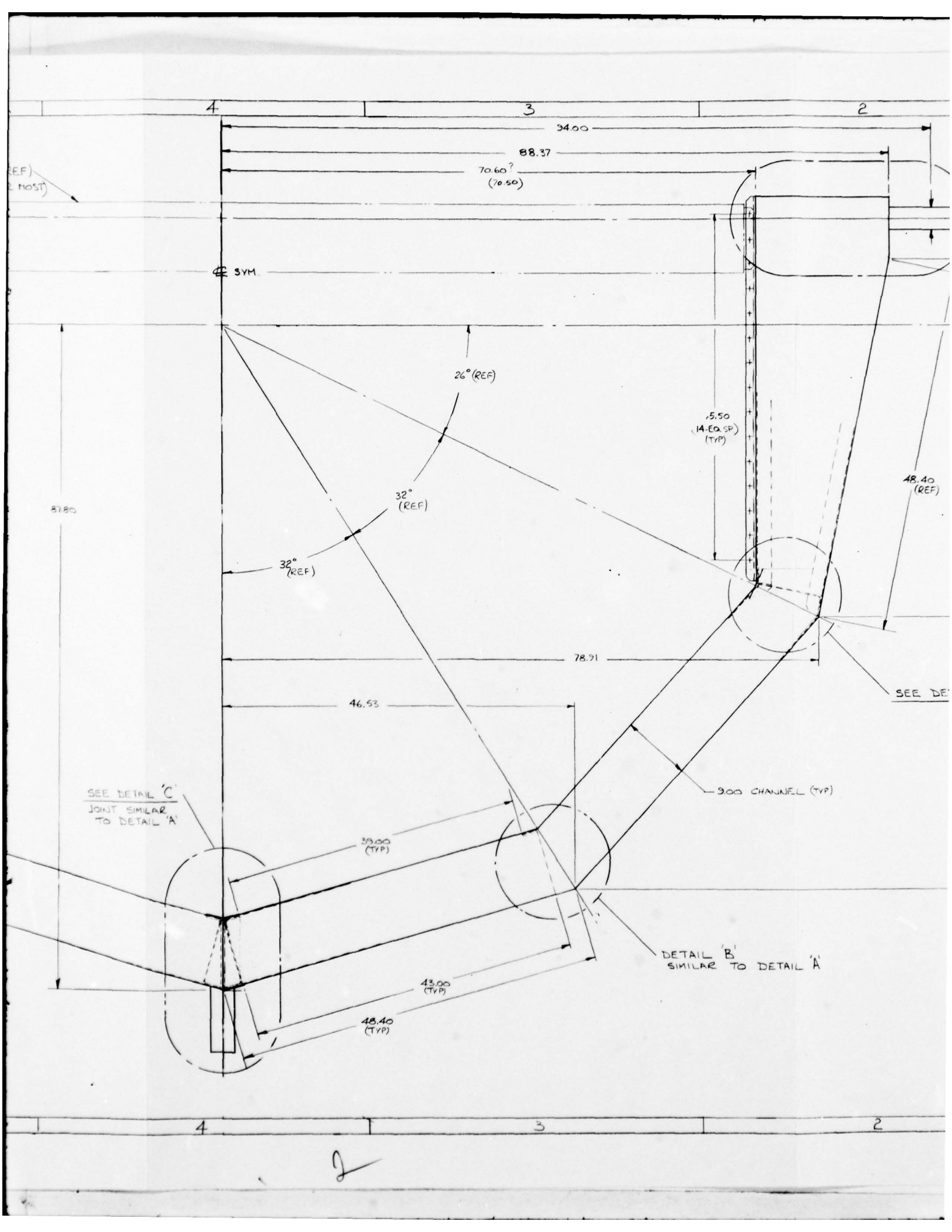
Hardware	Design Status	Make/ Buy	Source	Prior Use
Communication Subsystem				
Antennas	E	M	GE	BSE, DSCS III
Hybrid	E	B	Anaren	DSCS III
Diplexer	E	B	Wavecom	DSCS III
Transmitter	E	B	Cubic	DSCS III
Receiver	E	B	Cubic	DSCS III
Recorder	E	B	RCA	TIROS N, NIMBUS, SEASAT
TLM Unit	N	M	GE	-
CMD Decoder	N	M	GE	-
Decryptor	E	GFE	USAF	-
Encryptor	E	GFE	USAF	-
Payload I/F Unit	N	M	GE	-
Power Subsystem				
Power Controller	N	M	GE	-
Ordnance Cont.	M	M	GE	BSE
Battery	N	M	GE	-
Case	N	M	GE	-
Cells	M	B	Honeywell	-
Harness	N	M	GE	-
Structure Subsystem				
STR Frame	N	M	GE	-
Standard Panel	N	M	GE	-
Filler Panel	N	M	GE	-
Mechanisms				
Sep. Nuts	E	B	-	DSCS
Pin Puller	E	B	-	BSE
Marmon Clamp	E	B	-	Many
Thermal Control S/S				
Heat Pipes	M	B	-	-
Heat Exchanger	N	M	GFE	-
Blankets	M	M	GE	Many
Tapes/Coatings	E	B	Many	Many
Attitude Control Subsystem				
Sun Sensor	E	B	-	BSE
Inertial Reference Unit	E	B	-	MMS
Star Tracker	E	B	-	MMS
Gimbel Assembly	E	M	GE	-
Drive Electronics	N	M	GE	-
Power Conditioner	N	M	GE	-
Gyro	E	B	-	-
Optical Encoder	E	B	-	-
Computer	E	B	-	MMS
Power Conditioner	N	M	GE	-

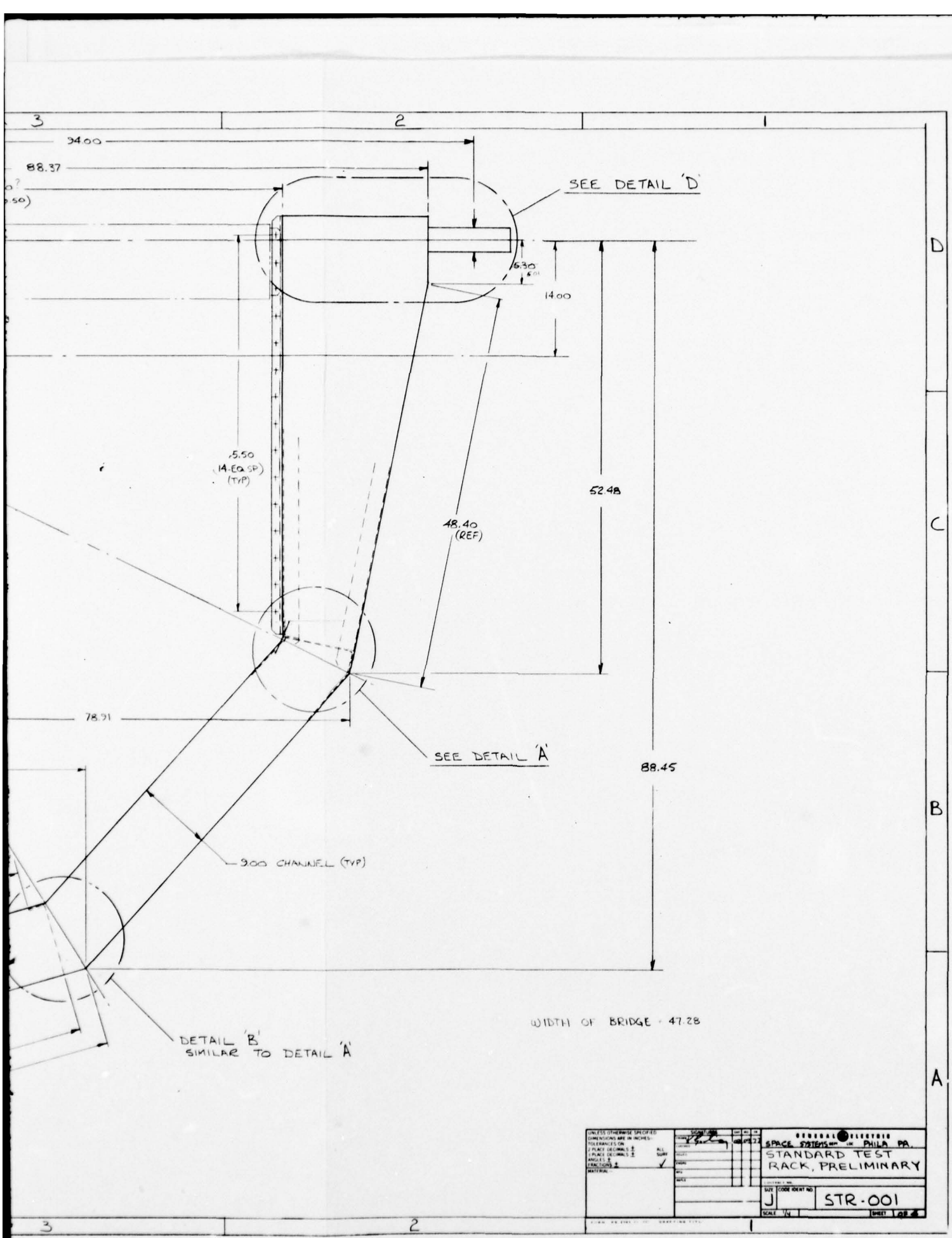
STR STUDY	HARDWARE USAGE										
Hardware	Qty Per STR	EM	EM S/S	SDM	Comp. Qual	Proto S/S	Proto Panel	Proto Comp. Accept	Proto STR Integ.	Flt Comp. Accept	Flt STR Intel.
Communication Subsystem											
Antennas	3			M3		-	-	aA3	aB3	bA3	bB3
Hybrid	1			M1		aB1	aC1	aA1	aD1	bA1	bB1
Diplexer	1			M1		aB1	aC1	aA1	aD1	bA1	bB1
Transmitter	4			M4		aB4	aC4	aA4	aD4	bA4	bB4
Receiver	2			M2		aB2	aC2	aA2	aD2	bA4	bB2
Recorder	4			M4		aB4	aC4	aA4	aD4	bA4	bB4
TLM Unit	1	aA1		M1	bA1	bB1	bC1	-	bD1	cA1	cB1
CMD Decoder	1	aA1		M1	bA1	bB1	bC1	-	bD1	cA1	cB1
Decryptor	2			M2		aA2	aB2	-	aC2	-	bA2
Encryptor	2			M2		aA2	aB2	-	aC2	-	bA2
Payload I/F Unit	1 (Mass Model / No. Required)	aA1		M1	bA1 (No. Required / Use (A = 1st, B = 2nd, etc.) / Item (a = 1st, b = 2nd, etc.))	bB1	bC1	-	bD1 (Typical)	cA1	cB1
Power Subsystem											
Power Controller	1			M1	aA1	-	aB1	-	aC1	bA1	bB1
Ordnance Cont.	1			M1	aA1	-	aB1	-	aC1	bA1	bB1
Battery	8/ Panel			M-	aA1	-	-	bA1	aB1 bB-	cA-	cB-
Case	1/Batt										
Cells	12/Batt										
Harness	1 Set			M1		aBp (Partial)	aCP	aA1	aD1	bA1	bB1
Structure Subsystem											
STR Frame	1			aA1	-	-	-	-	aB1	bA1	bB1
Standard Panel	8			aA8	-	aB2	aC1	-	aB6 aC1 aD1	bA8	bB9
Filler Panel	1			aA1	-	-	-	-	aB1	bA1	bB1
Mechanisms											
Sep. Nuts	-			aA-	-	-	-	bA-	bB-	cA-	cB-
Pin Pullers	-			aA-	-	-	-	bA-	bB-	cA-	cB-
Marmon Clamp	-			aA-	-	-	-	bA-	bB-	cA-	cB-
Thermal Control Subsystem											
Heat Pipes	4/Panel 12/STR Frame			M-	aA1	-	-	bA-	bB-	cA-	cB-
Heat Exchanger	1	aA1		M1	bA1	-	-	-	bB1	cA1	cB1
Blankets	AR			-	-	-	-	-	1 set	-	1 set
Tapes/Coatings	AR (As Required)			-	-	-	-	-	AR	-	AR
Attitude Subsystem Control											
Sun Sensor	2			M2		aB2	aC2	aA2	aD2	bA2	bB2
Inertial Reference Unit	1			M1		aB1	aC1	aA1	aD1	bA1	bB1
Star Tracker	2			M2		aB2	aC2	aA2	aD2	bA2	bB2
Gimbal Assembly	1			M1		aB1	aC1	aA1	aD1	bA1	bB1
Drive Electronics	1	aA1		M1	bA1	bB1	bC1	-	bD1	cA1	cB1
Power Conditioner	1	aA1		M1	bA1	bB1	bC1	-	bD1	cA1	cB1
Gyro	1			M1		aB1	aC1	aA1	aD1	bA1	bB1
Optical Encoder	2			M2		aB2	aC2	aA2	aD2	bA2	bB2
Computer	1			M1		aB1	aC1	2A1	aD1	bA1	cB1
Power Conditioner	1	aA1		M1	bA1	bB1	bC1	-	bD1	cA1	cB1

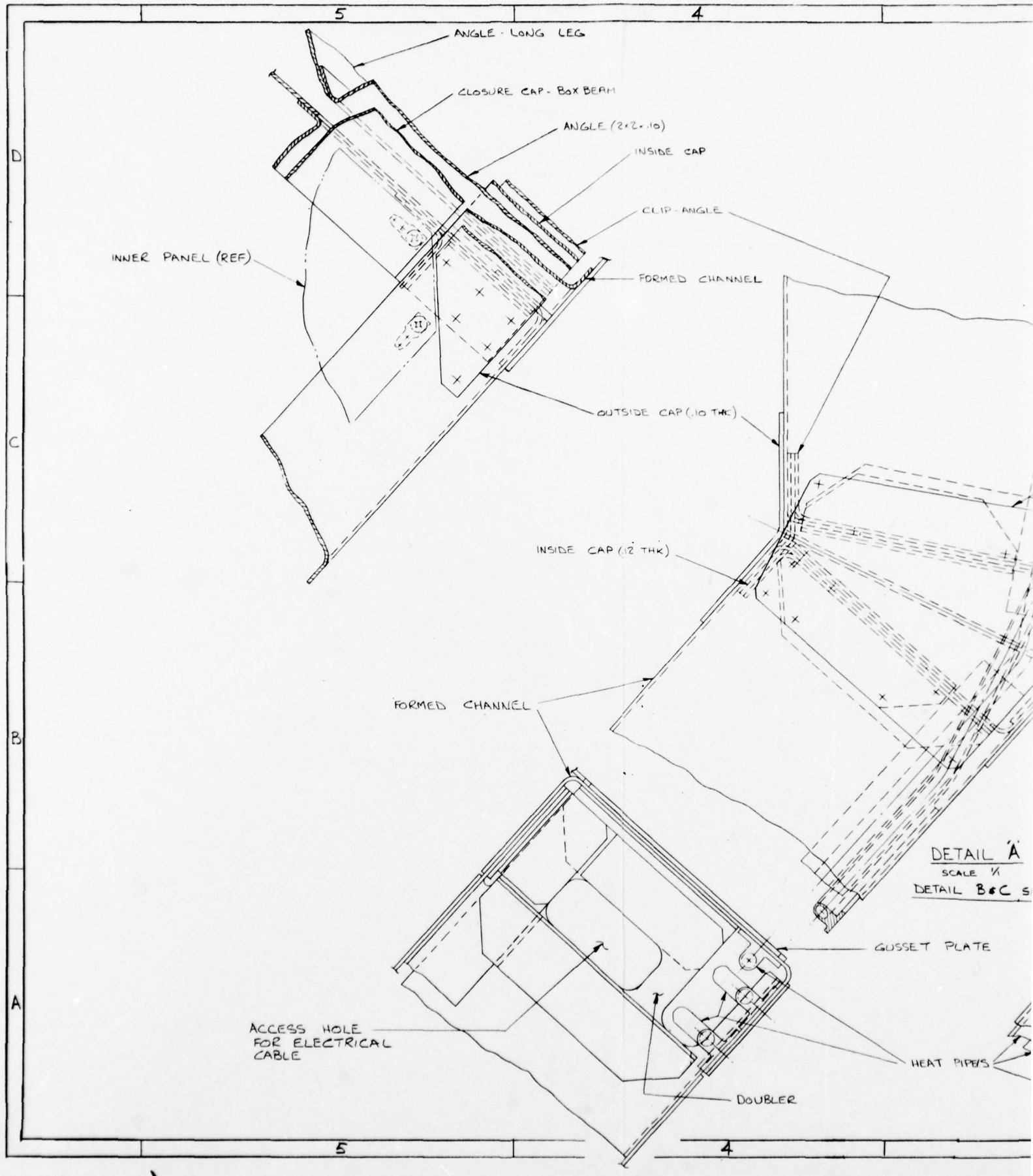
APPENDIX E

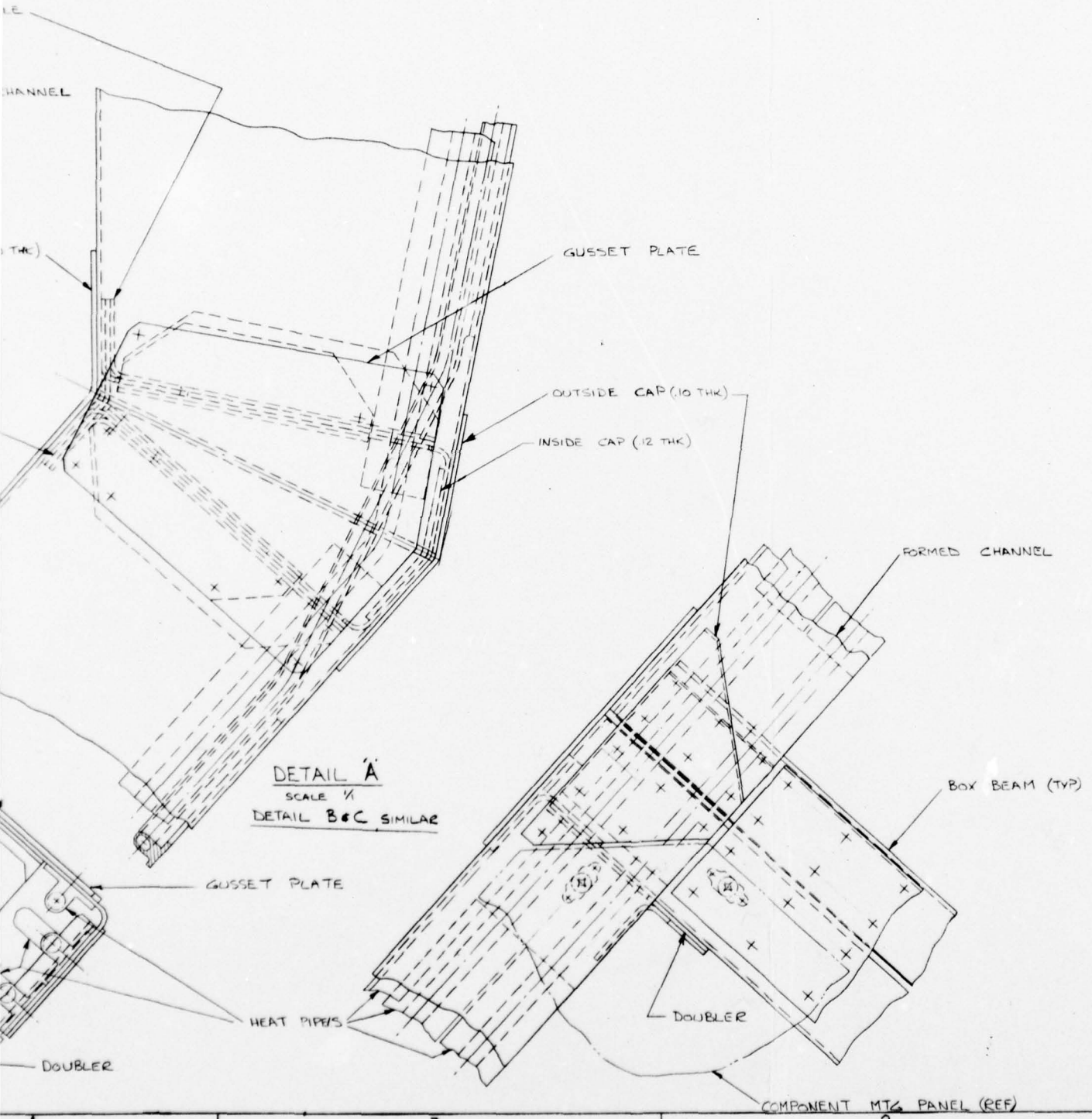
DESIGN DRAWINGS


6
5
4
D
C
B
A
BRIDGE (REF)
(SHOWN IN UPPER MOST)
POSITION
87.80
SEE DETAIL 'C'
JOINT SIMILAR
TO DETAIL 'A'

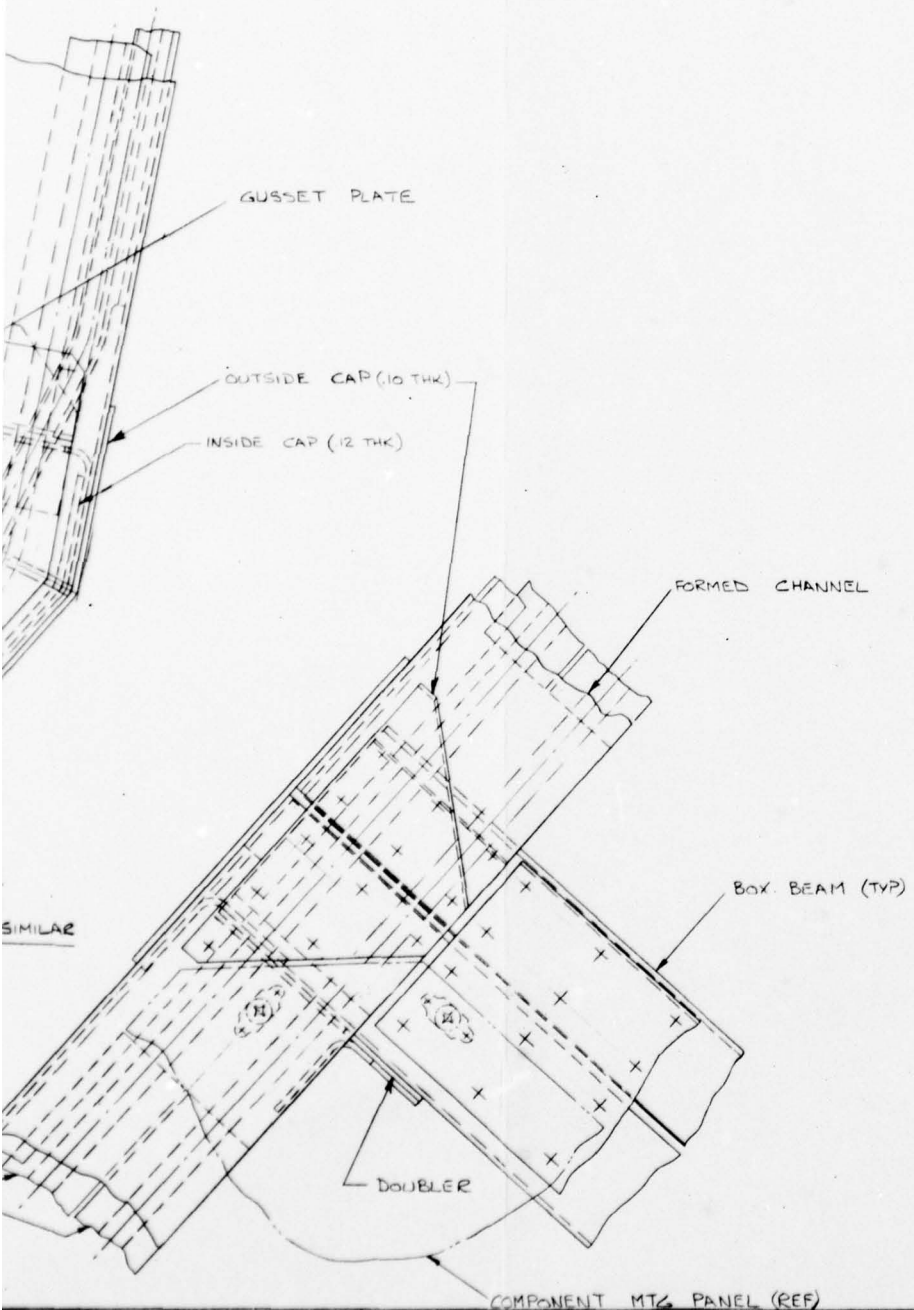

4
3
2
34.00
88.37
70.60?
(70.50)
EF)
MOST)
℄ SYM.
26° (REF)
32°
(REF)
32°
(REF)
87.80
5.50
14 EQ SP
(TYP)
48.40
(REF)
78.91
SEE DE
46.53
SEE DETAIL 'C'
JOINT SIMILAR
TO DETAIL 'A'
9.00 CHANNEL (TYP)
39.00
(TYP)
DETAIL 'B'
SIMILAR TO DETAIL 'A'
43.00
(TYP)
48.40
(TYP)
4
3
2
2


34.00
88.37
SEE DETAIL 'D'
6.30
14.00
5.50
(14 EQ SP)
(TYP)
52.48
48.40
(REF)
78.91
SEE DETAIL 'A'
88.45
9.00 CHANNEL (TYP)
WIDTH OF BRIDGE = 47.28
DETAIL 'B'
SIMILAR TO DETAIL 'A'
GENERAL ELECTRIC
SPACE SYSTEMS
PHILA PA.
STANDARD TEST
RACK, PRELIMINARY
SIZE J
STR-001
SCALE 1/4
SHEET


5
4
D
C
B
A
ANGLE - LONG LEG
CLOSURE CAP - BOX BEAM
ANGLE (2×2×.10)
INSIDE CAP
CLIP - ANGLE
INNER PANEL (REF)
FORMED CHANNEL
OUTSIDE CAP (.10 THK)
INSIDE CAP (.12 THK)
FORMED CHANNEL
DETAIL A
SCALE 1/1
DETAIL B&C S
GUSSET PLATE
ACCESS HOLE FOR ELECTRICAL CABLE
HEAT PIPES
DOUBLER
5
4



DETAIL A
SCALE 1/1
DETAIL B & C SIMILAR



SIMILAR

UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES
TOLERANCES ON
2 PLACE DECIMALS ± ALL SURF
3 PLACE DECIMALS ±
ANGLES ±
FRACTIONS ± ✓
MATERIAL —

SIGNATURES

DRAWN
CHECKED
ENGR
MFG
QUALITY

DATE 10 10 77

GENERAL ELECTRIC
SPACE SYSTEMS DIV. VAL. PHILA. PA.

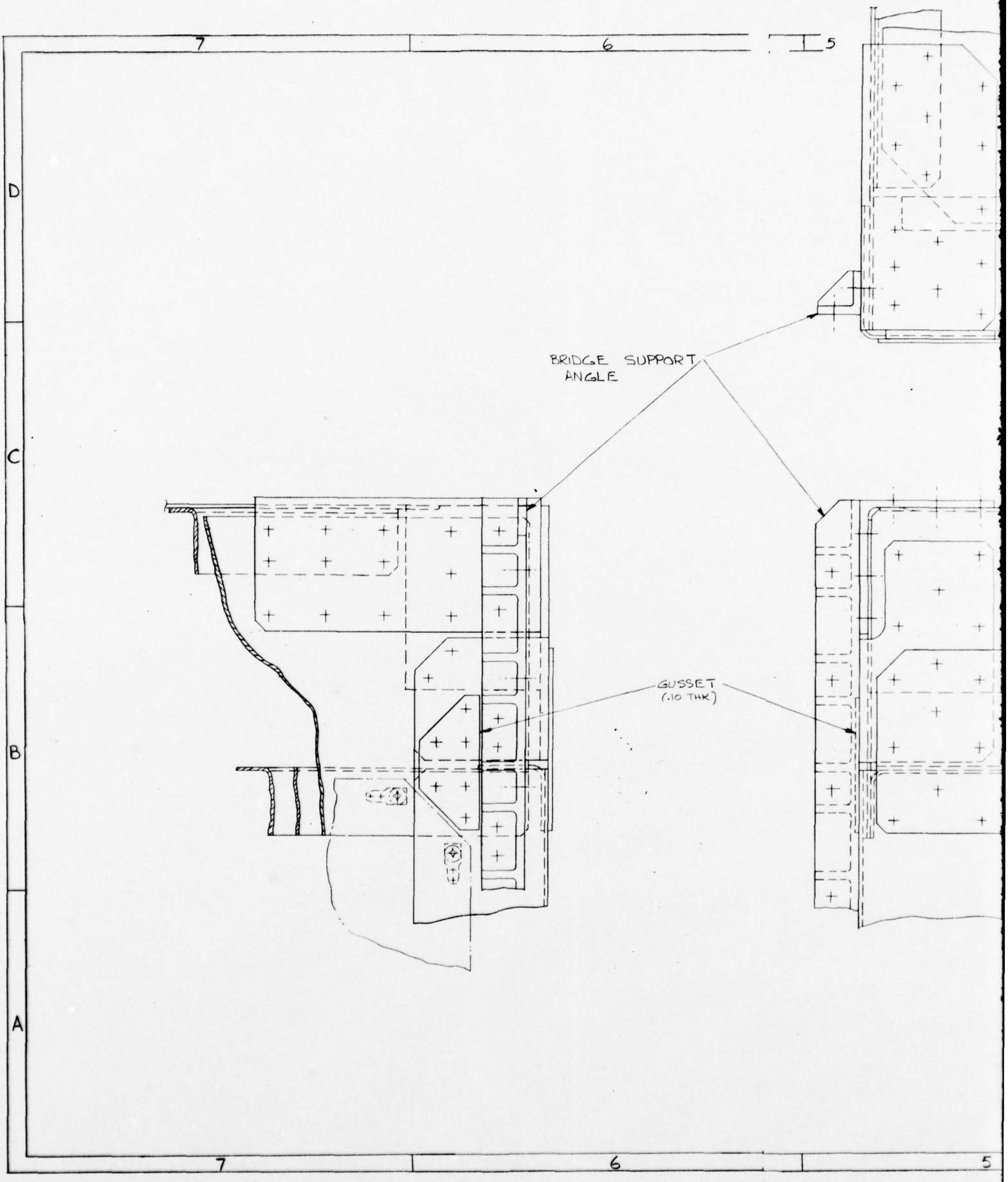
STANDARD TEST RACK
PRELIMINARY

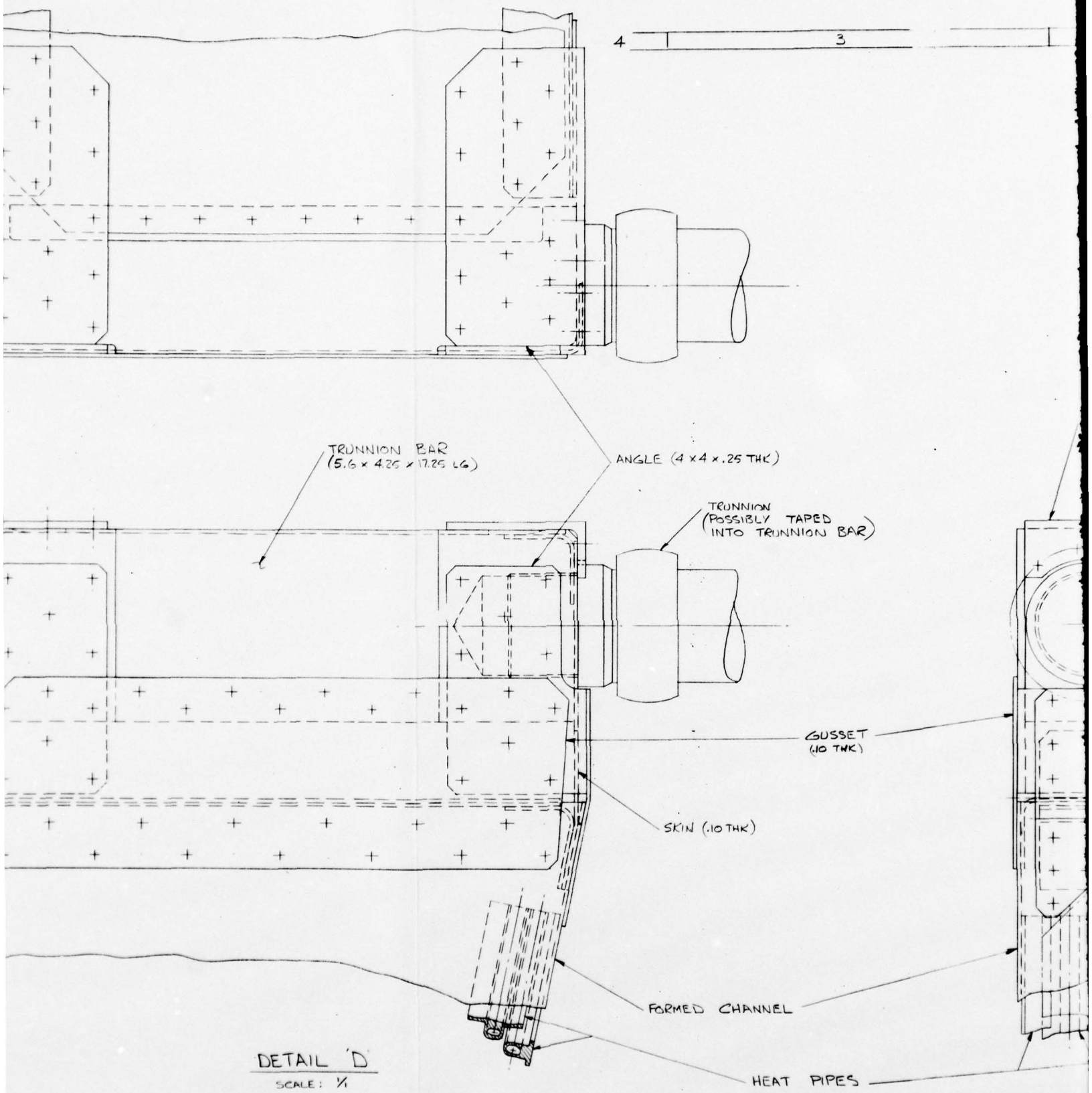
SIZE J | CODE IDENT NO. | STR-001

SCALE 1/4 | SHEET 2 of 6

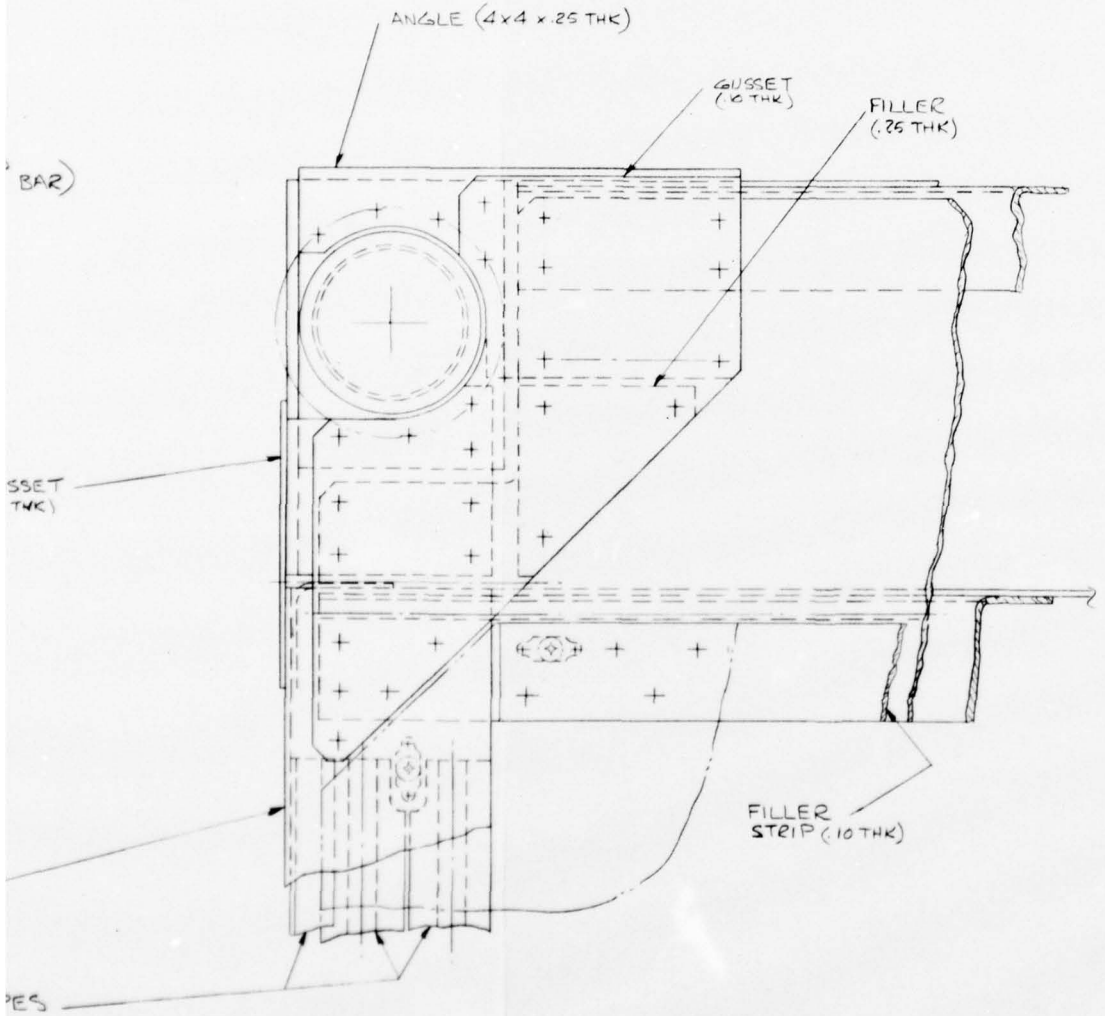
2

3

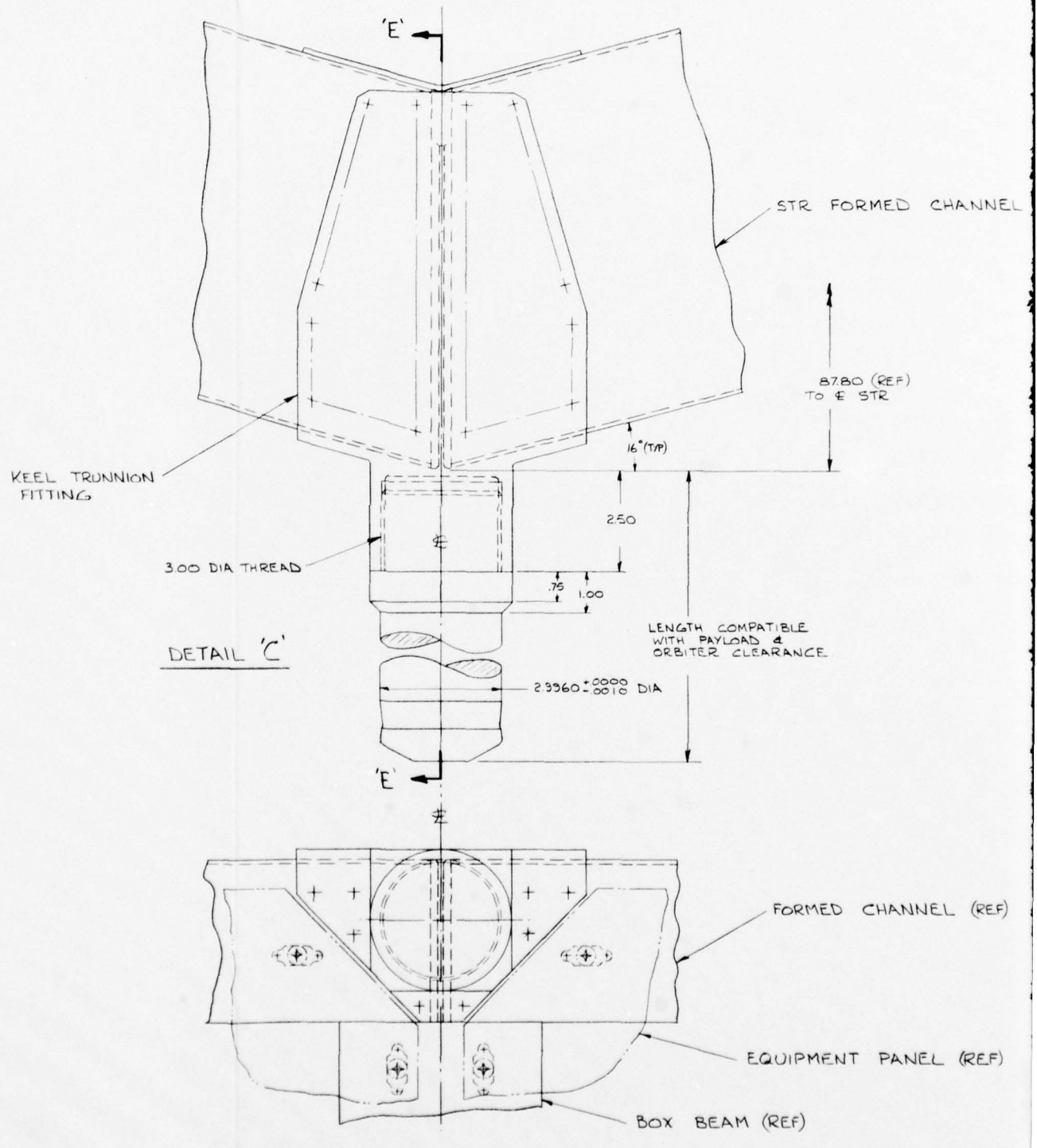

7
6
5
D
C
B
A
BRIDGE SUPPORT ANGLE
GUSSET (.10 THK)

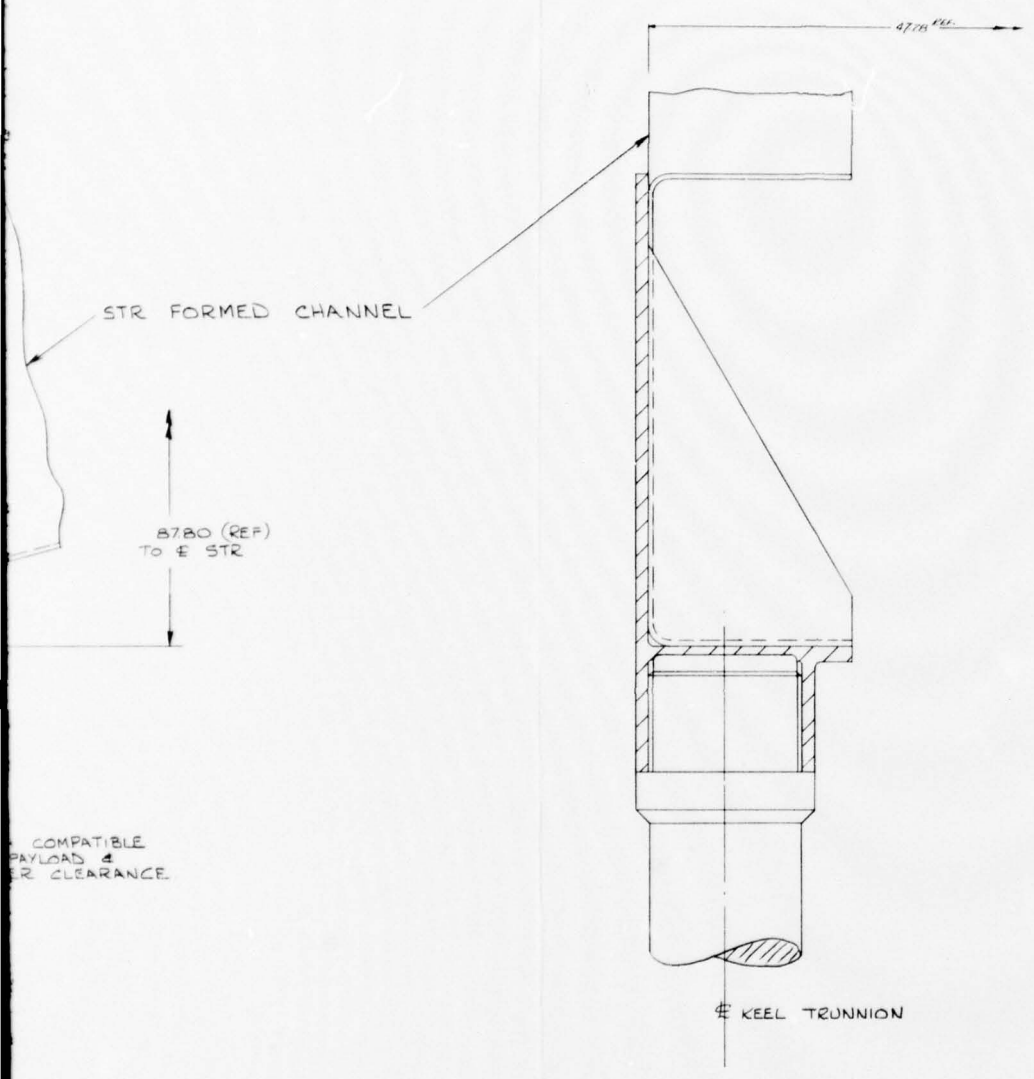


DETAIL 'D'
SCALE: 1/1



GENERAL ELECTRIC SPACE SYSTEMS DIV. VAL PHILA PA		
STANDARD TEST RACK PRELIMINARY		
SIZE J	CODE IDENT NO	STR - 001
SCALE		SHEET 1 OF 4


'E'
STR FORMED CHANNEL
87.80 (REF)
TO ℄ STR
16° (TYP)
KEEL TRUNNION
FITTING
2.50
3.00 DIA THREAD
.75
1.00
LENGTH COMPATIBLE
WITH PAYLOAD &
ORBITER CLEARANCE
DETAIL 'C'
2.3960 +.0000 -.0010 DIA
'E'
FORMED CHANNEL (REF)
EQUIPMENT PANEL (REF)
BOX BEAM (REF)



SECT 'E-E'

COMPATIBLE
PAYLOAD &
ER CLEARANCE

FORMED CHANNEL (REF)

EQUIPMENT PANEL (REF)

BEAM (REF)

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES— TOLERANCES ON:	SIGNATURES		
2 PLACE DECIMALS ± ALL	DRAWN	15	10 77
3 PLACE DECIMALS ± SURF	CHECKED		
ANGLES ± ✓	ENGR		
FRACTIONS ±	MFG		
MATERIAL —	QUALITY		
	DATE		

GENERAL ELECTRIC
SPACE SYSTEMS DIV. VALLEY FORGE, PA.

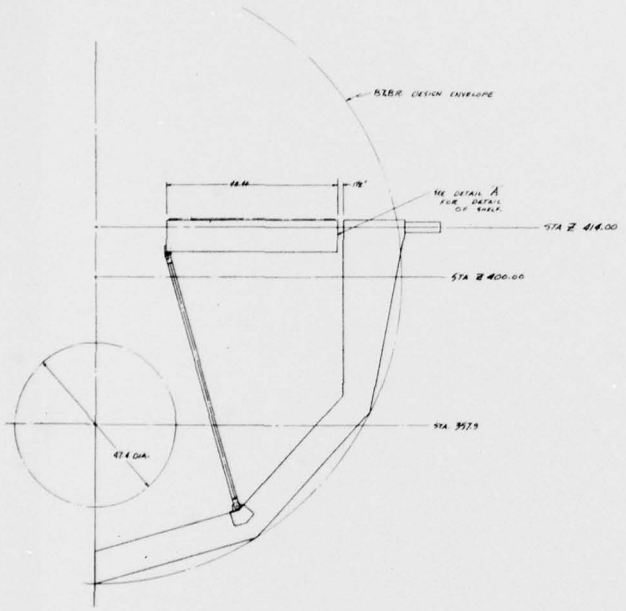
STANDARD TEST RACK
PRELIMINARY

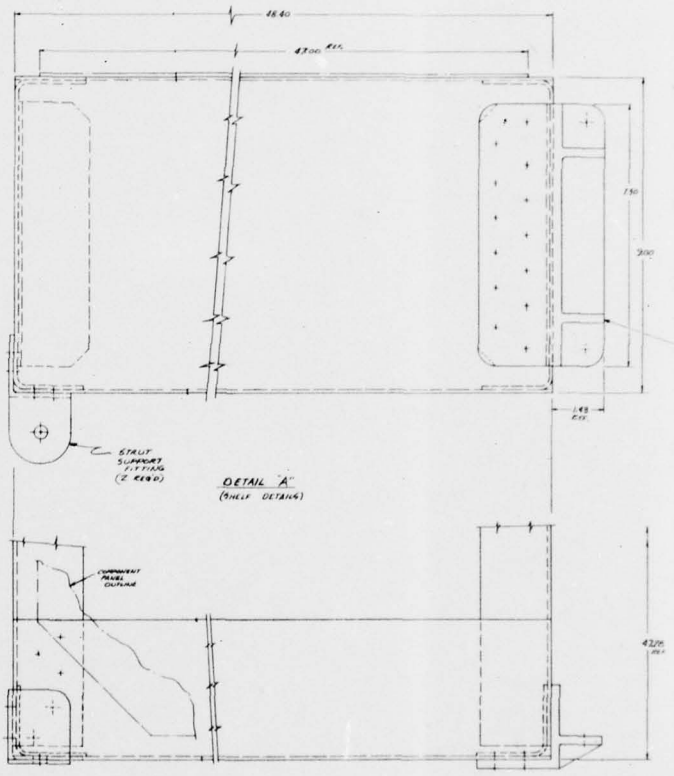
CONTRACT NO.

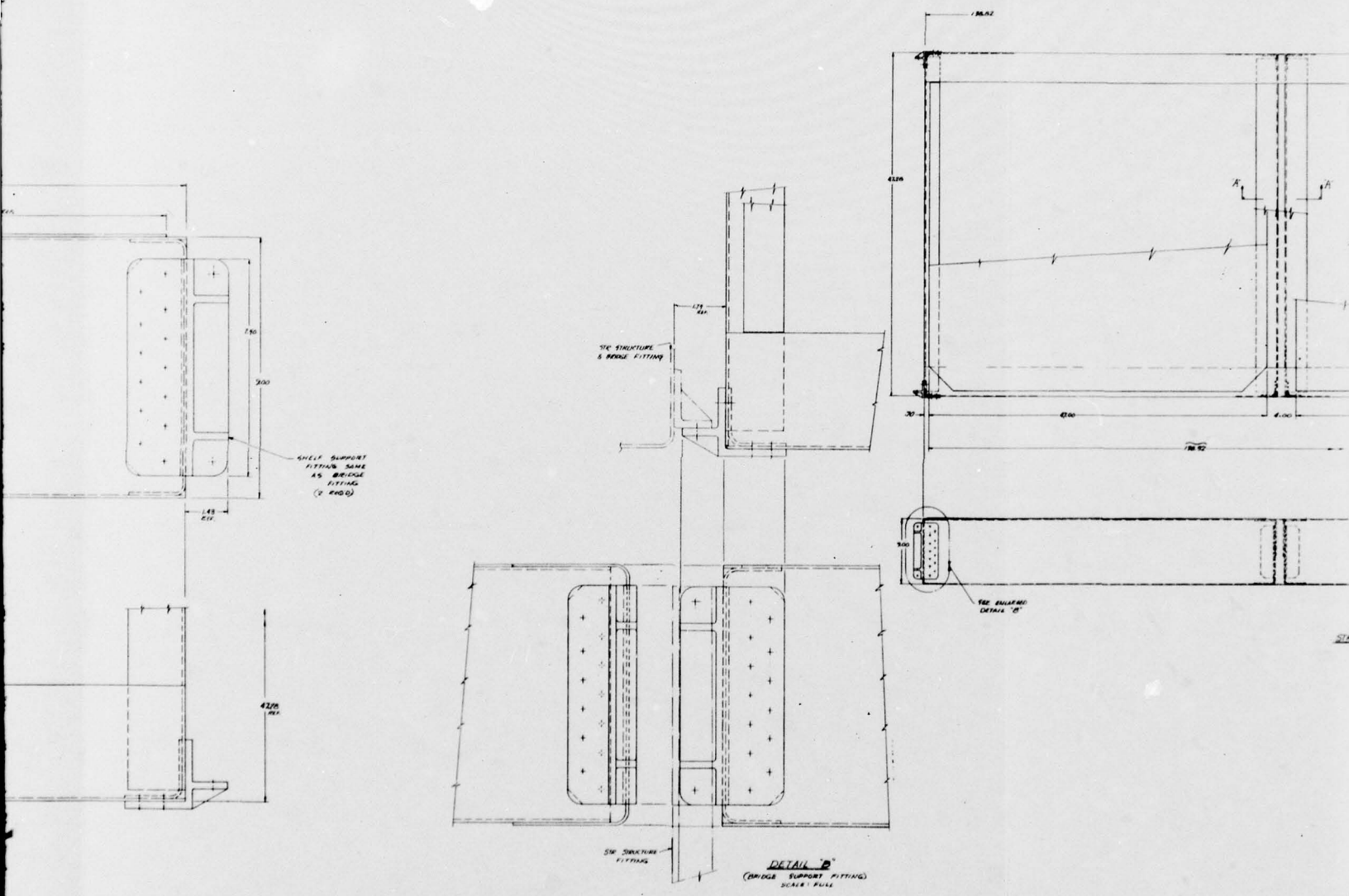
SIZE	CODE IDENT NO.	
J		STR-001

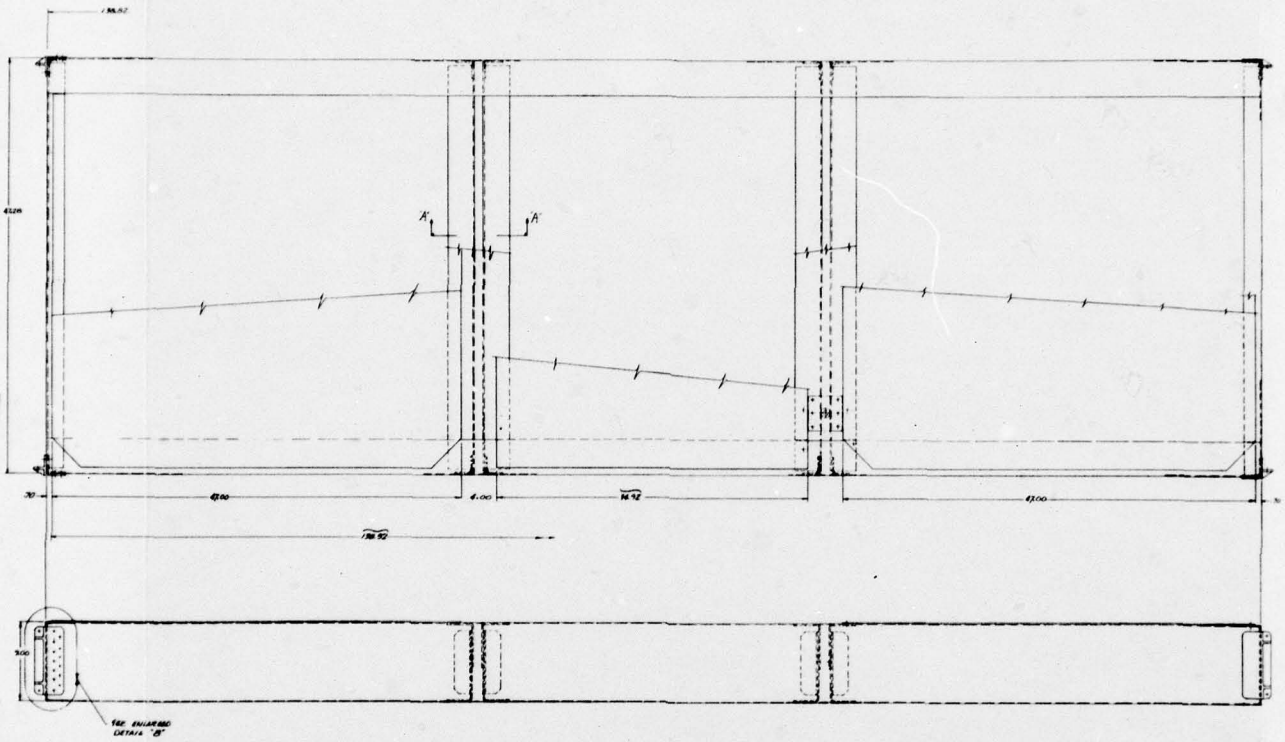
SCALE 1/1 SHEET 4 OF 4

2


B2BK DESIGN ENVELOPE
SEE DETAIL A FOR DETAIL OF SHELF
STA Z 414.00
STA Z 400.00
STA 357.9
47.4 DIA.


48.40
47.00 REF.
STRUT SUPPORT FITTING (Z 400)
DETAIL "A"
(SHELF DETAIL)
COMPARTMENT PANEL OUTLINE

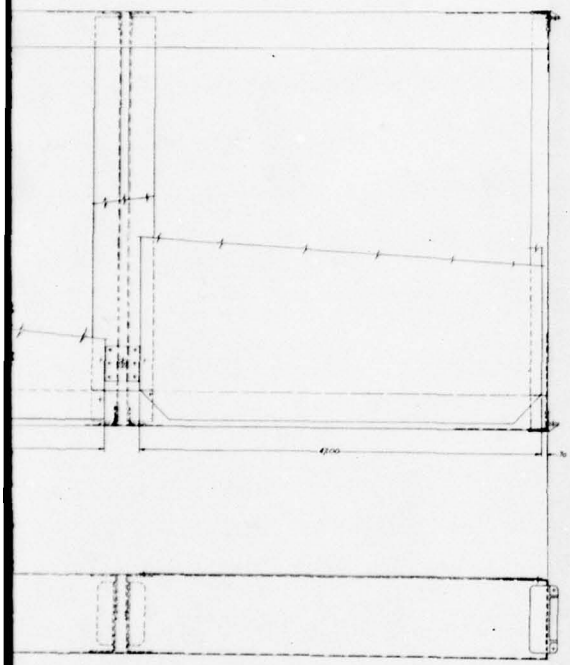

SHELF SUPPORT FITTING SAME AS BRIDGE FITTING (2 REQ'D)
STR STRUCTURE & BRIDGE FITTING
STR STRUCTURE FITTING
SEE BULKHEAD DETAIL 'B'
DETAIL 'B'
(BRIDGE SUPPORT FITTING)
SCALE 1 FULL



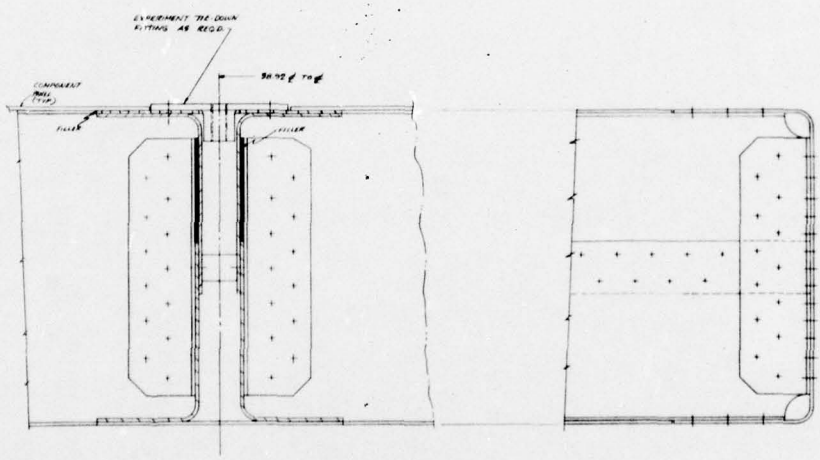
STR BRIDGE STRUCTURE DETAILS

SCALE 1:X

DETAIL "B"
SUPPORT FITTING
SCALE: FULL



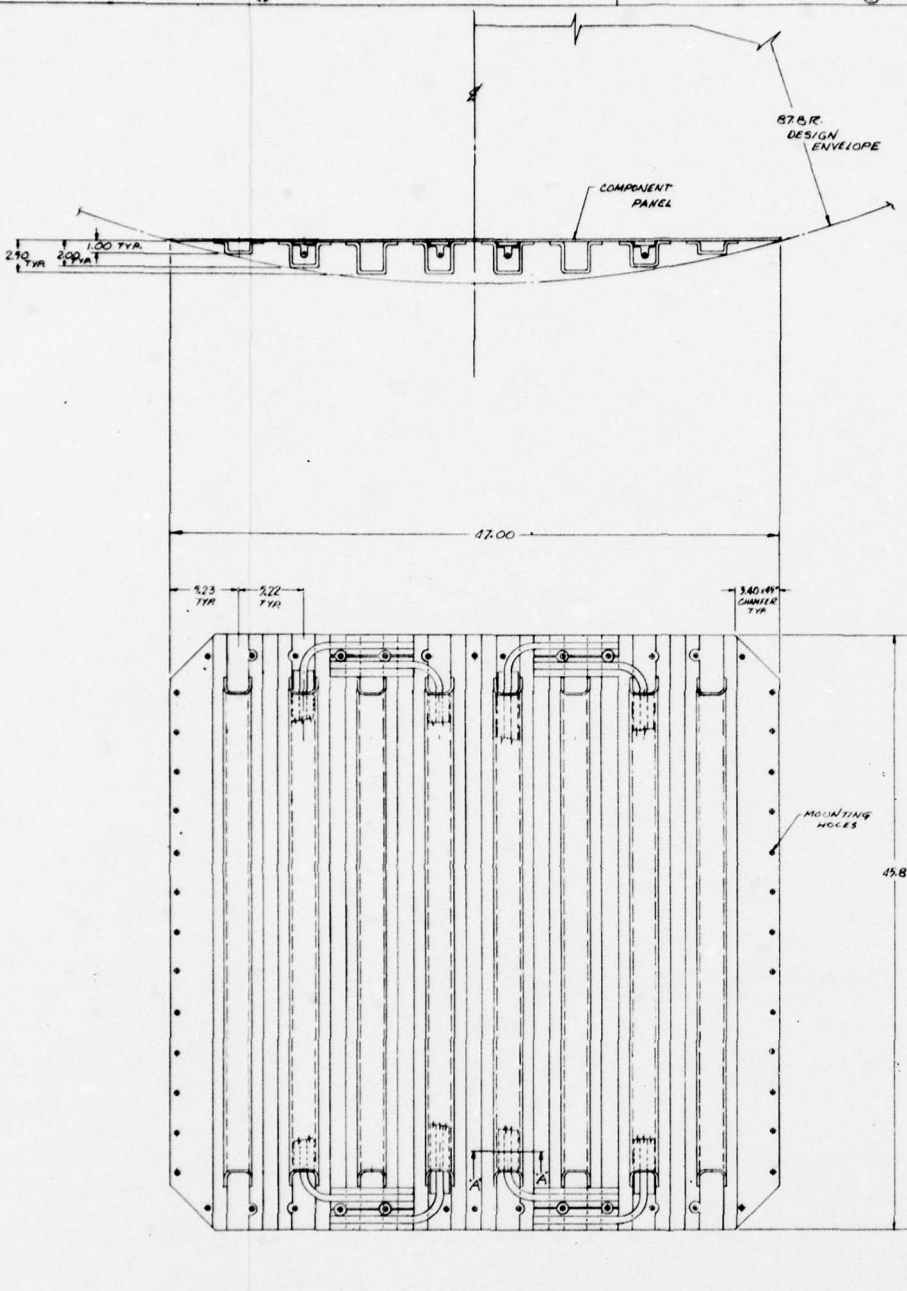
DETAILS



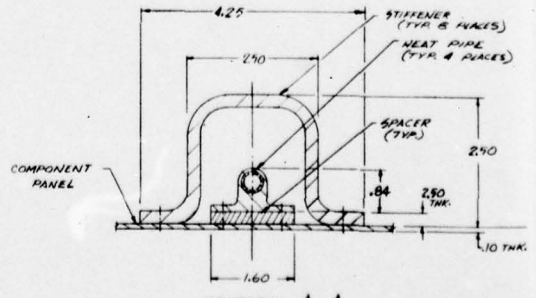
SECTION "A-A"
SCALE: FULL

			STR'L SHELF AND BRIDGE STRUCTURE CONFIGURATION
			STR-002

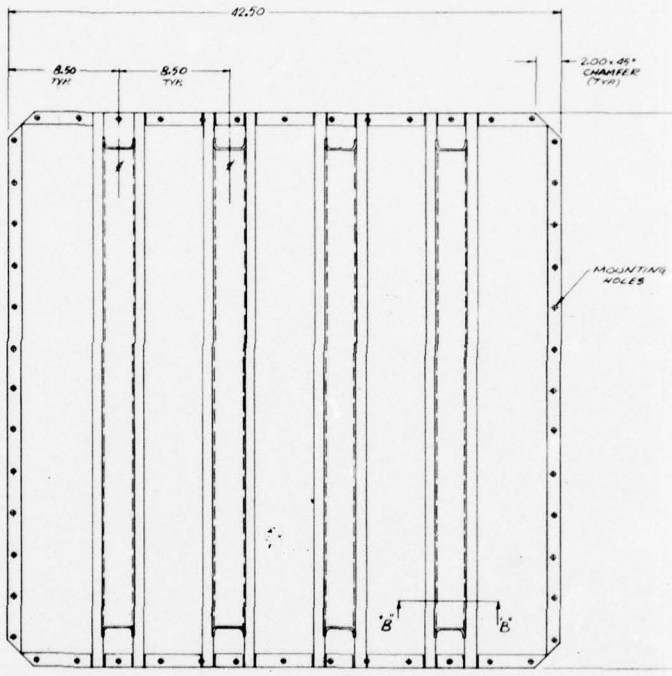
4



COMPONENT PANEL DETAIL
SCALE - 1/4

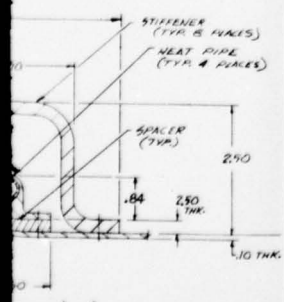


SECTION A-A
SCALE: FULL

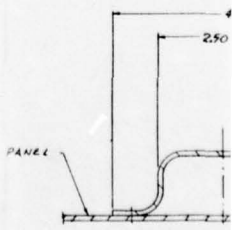


INNER PANEL DETAIL
(LOOKING INBOARD)
SCALE 1/4"



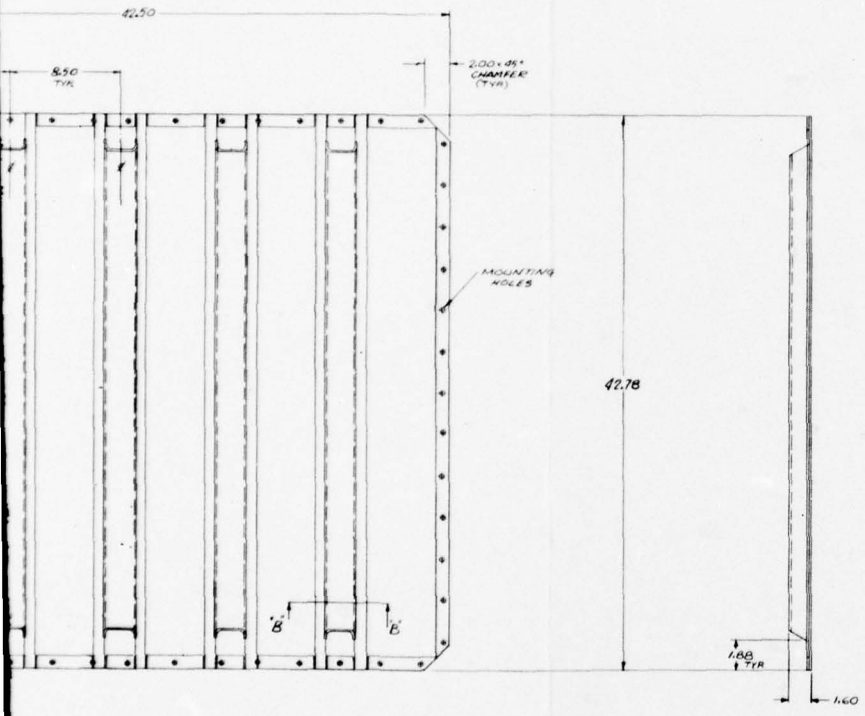


ON A-A
LE: FULL

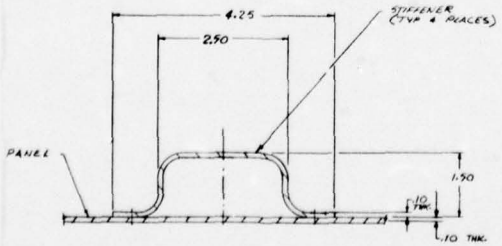


SECTION
SCALE

J



INNER PANEL DETAIL
(LOOKING INBOARD)
SCALE 1/4"



SECTION "B-B"
SCALE - FULL

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES— TOLERANCES ON	SIGNATURES
2 PLACE DECIMALS ±	N.I.
3 PLACE DECIMALS ±	SURF
ANGLES ±	✓
FRACTIONS ±	
MATERIAL	

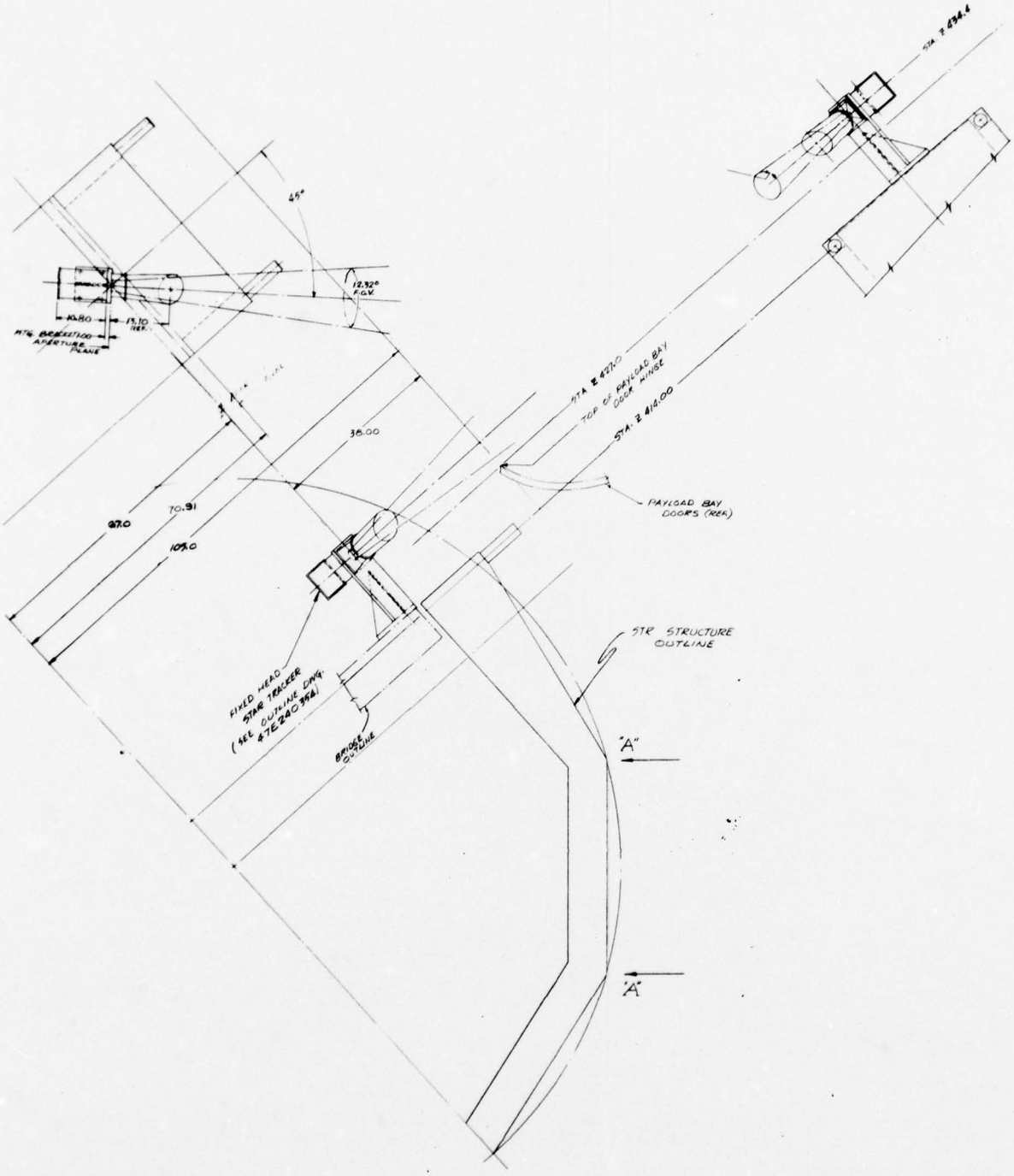
GENERAL ELECTRIC
S.S. DEPT. — PHILA. PA.

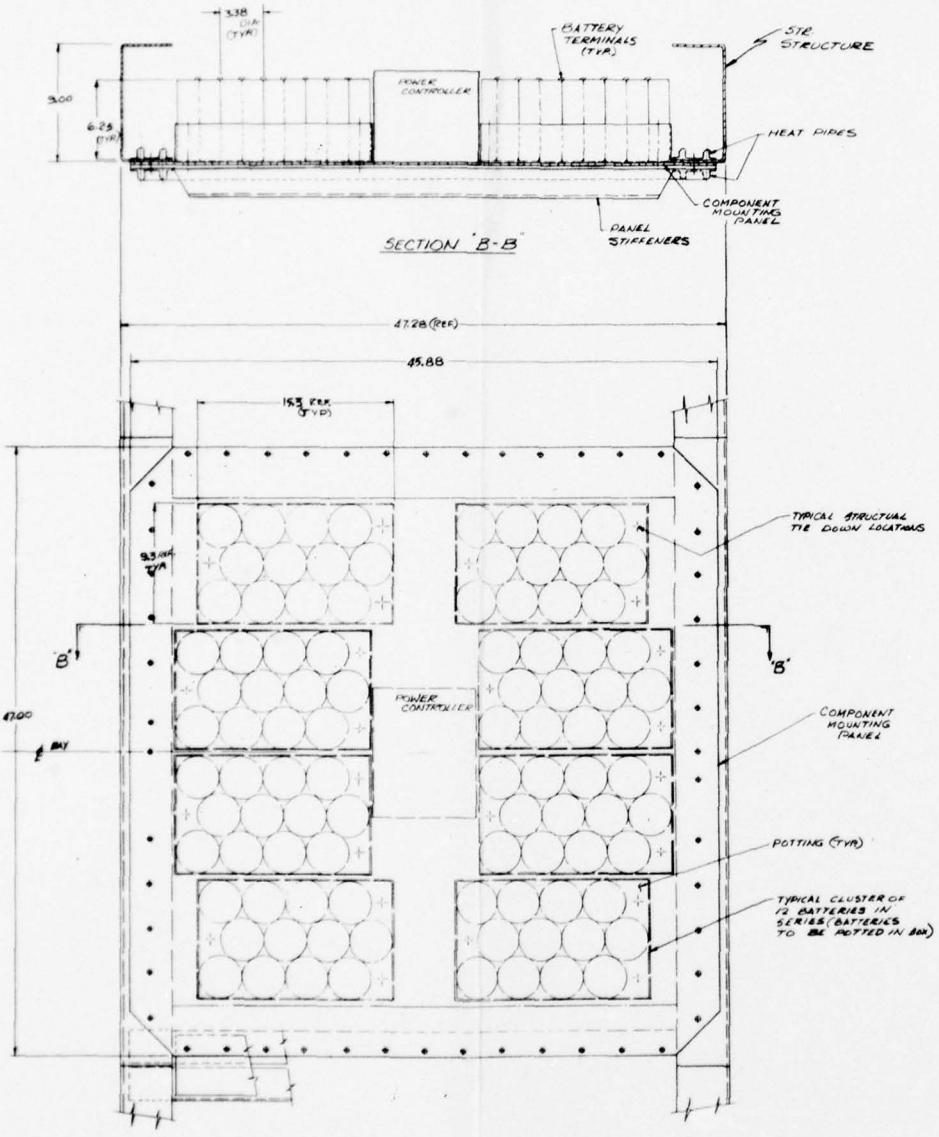
COMPONENT PANEL CONFIGURATIONS

SIZE J | CODE IDENT NO. | STR-003

SCALE | SHEET 1 OF 1

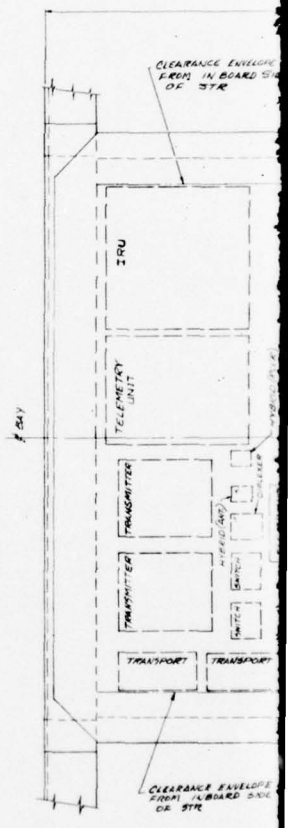
3

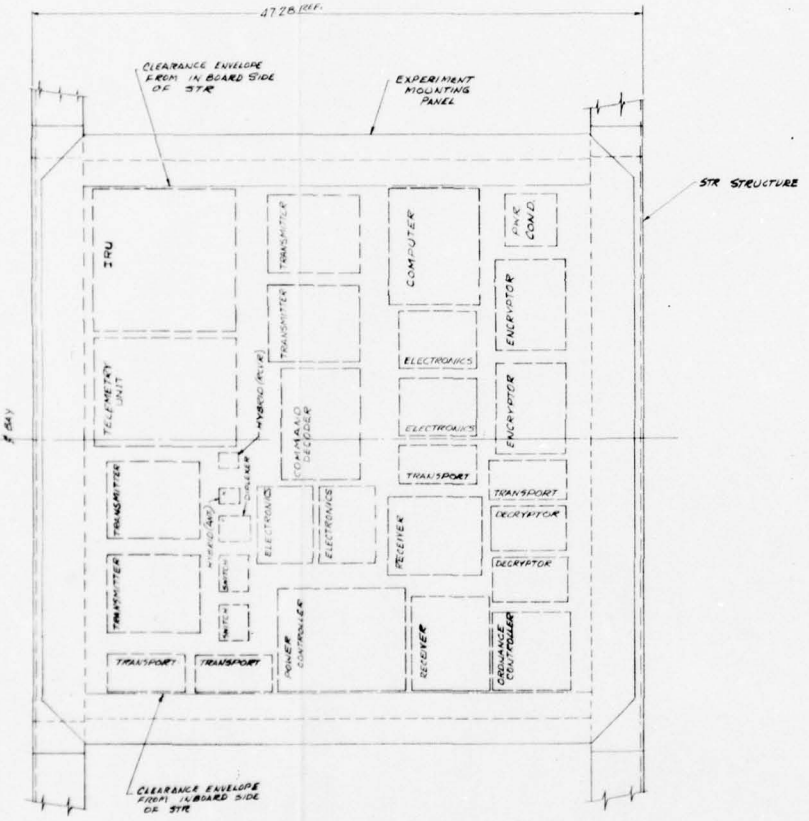

45°
12.32° F.O.V.
10.80
13.10 REF.
MTG. BRKT/TOD APERTURE PLANE
38.00
70.91
67.0
103.0
STA Z 427.0
TOP OF PAYLOAD BAY DOOR HINGE
STA Z 414.00
PAYLOAD BAY DOORS (REF.)
FIXED HEAD STAR TRACKER (SEE OUTLINE DWG. 4-7E240 9561)
BRIDGE OUTLINE
STR STRUCTURE OUTLINE
"A"
"A"
STA Z 432.4
3.00
B"
47.00
℄ BAY



SECTION 'B-B'

VIEW - A - A
(BATTERY BAY CONFIGURATION)
SCALE 1/4





VIEW - A - A
(ELECTRONIC BAY CONFIGURATION)
SCALE 1/4

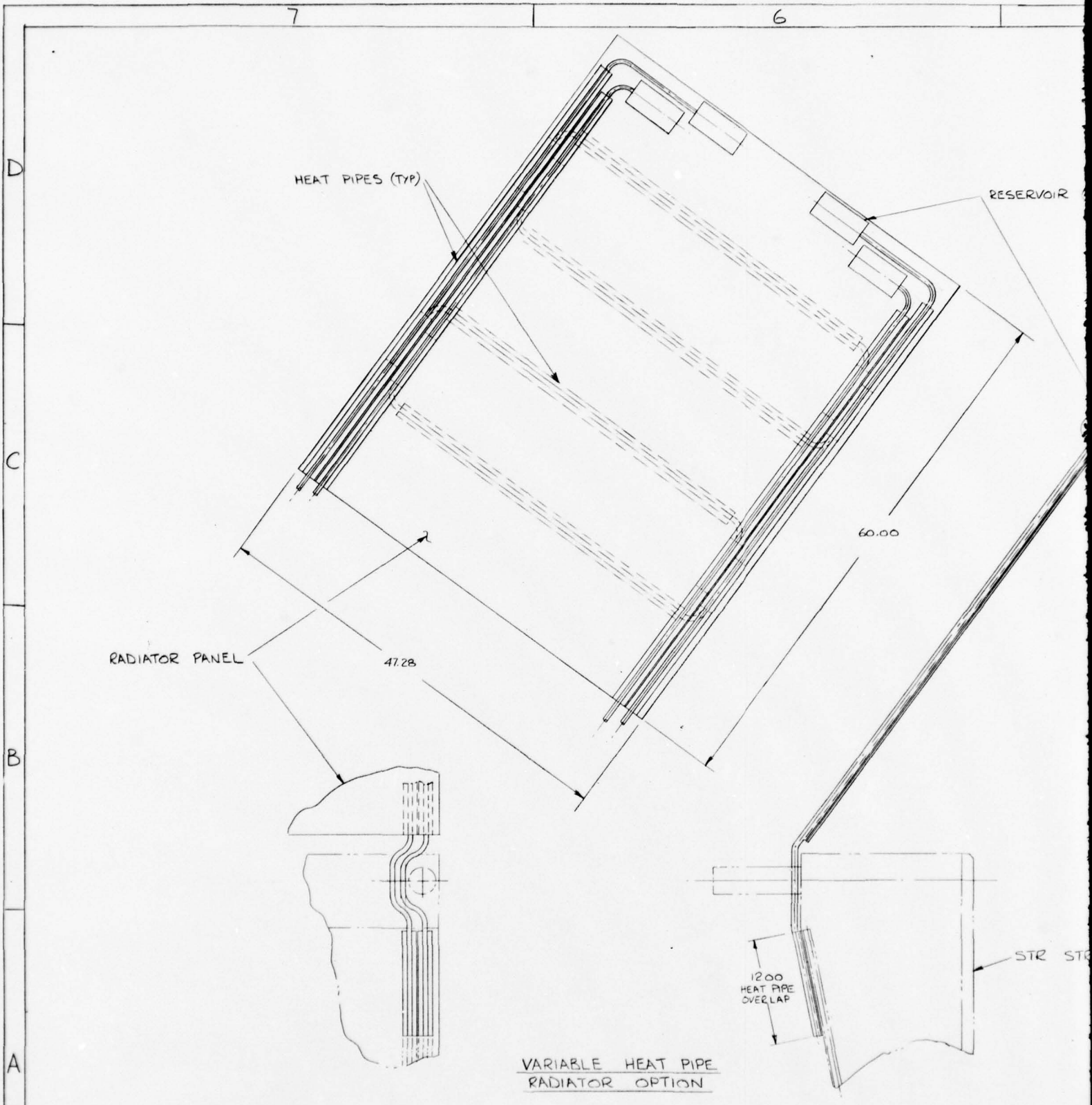
UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES ON
2 PLACE DECIMALS ± ALL SURF
3 PLACE DECIMALS ±
ANG. ±
FRACT. ±
MATERIAL

SIGNATURES — DRAWN Y. E. Ollis 12/18/77

GENERAL ELECTRIC
S.S. DEPT. VALLEY FORGE, PA.

COMPONENT PANEL CONFIGURATIONS

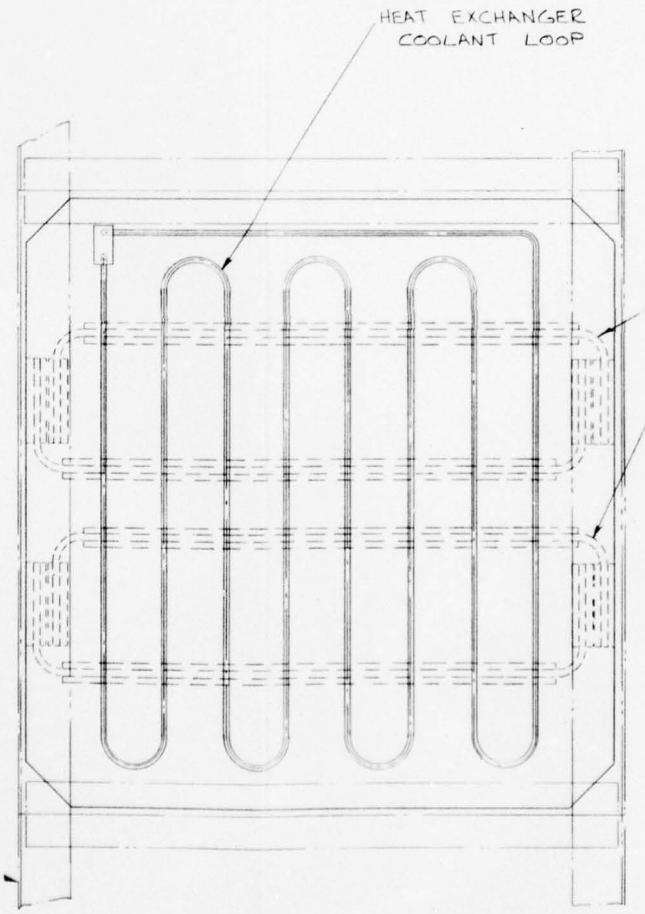
CONTRACT NO.
SIZE | CODE IDENT NO. | STR-003
SCALE 1/10 | SHEET 1 OF 3



VARIABLE HEAT PIPE
RADIATOR OPTION

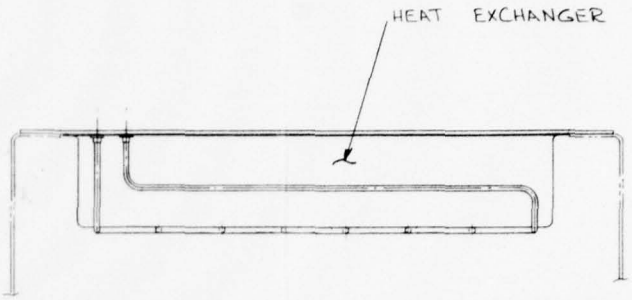


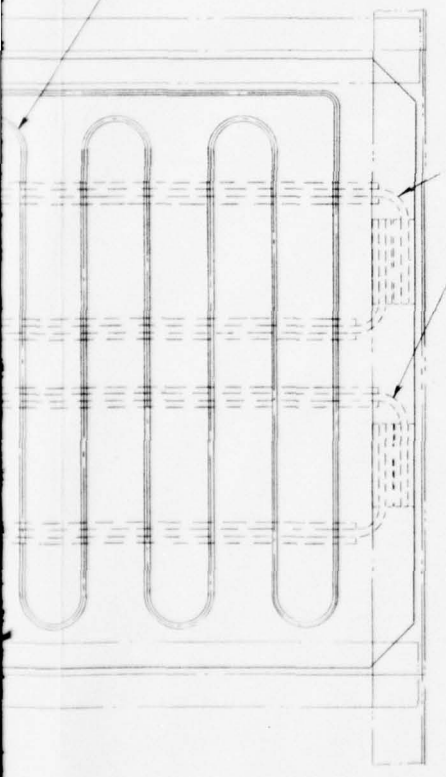




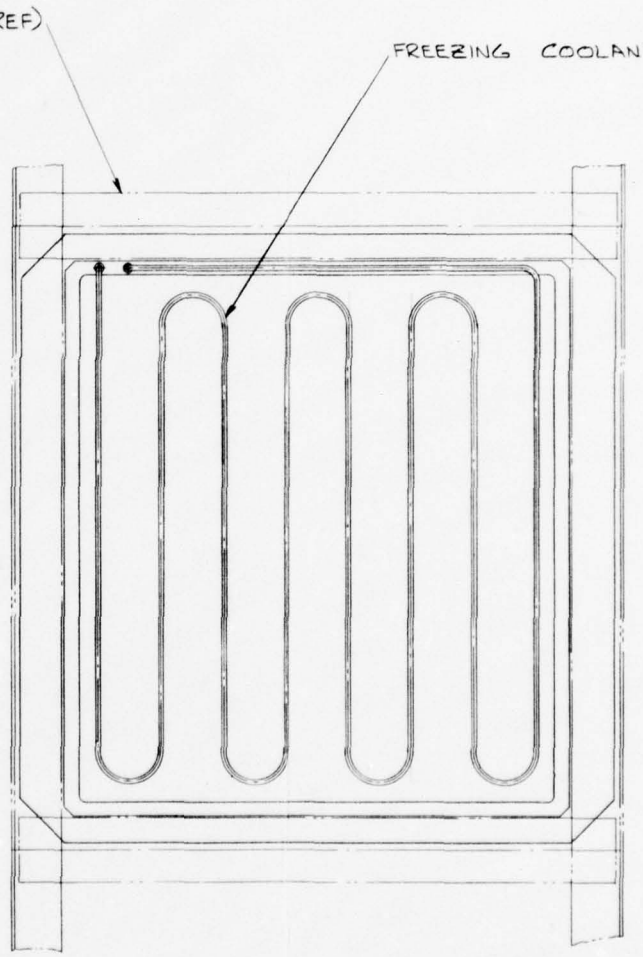
COLD PLATE
SHUTTLE HEAT EXCHANGER OPTION



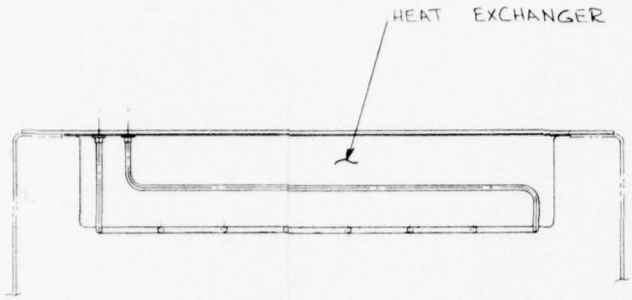


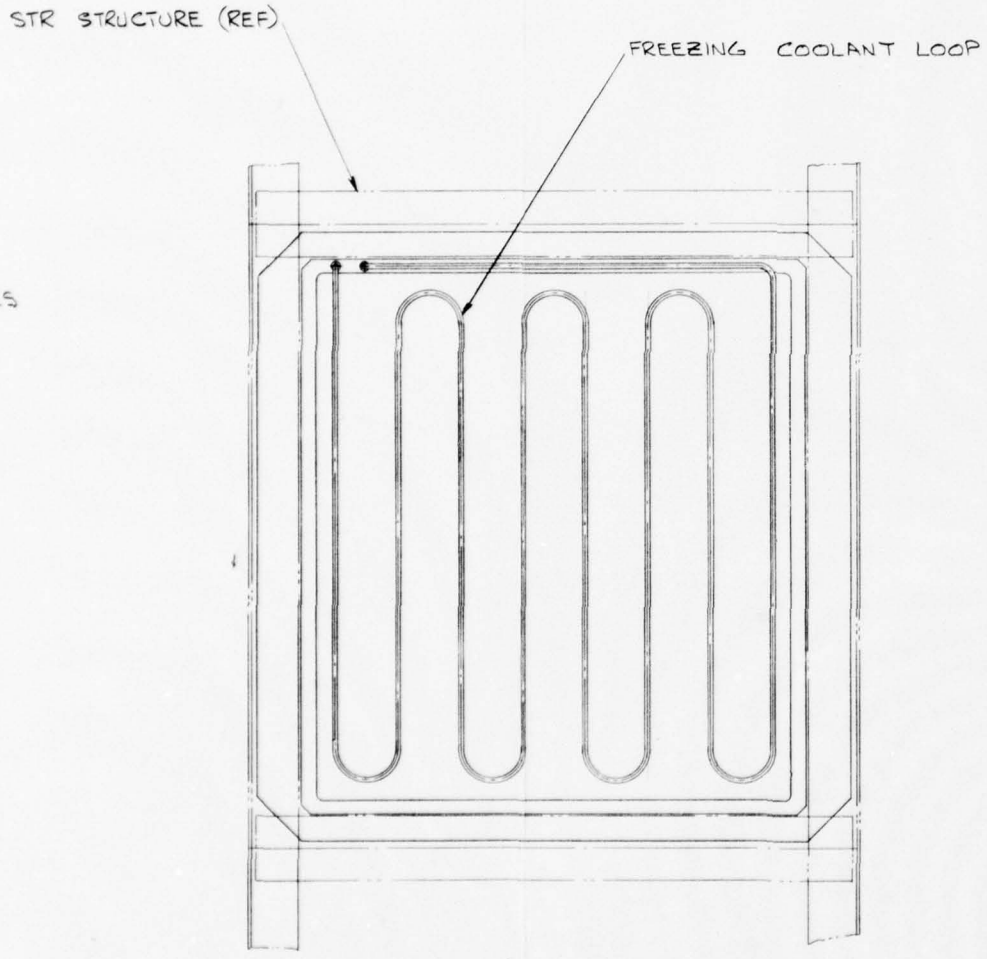


COLD PLATE
TTLE HEAT EXCHANGER OPTION



CHANGE OF PHASE
MODULE OPTION

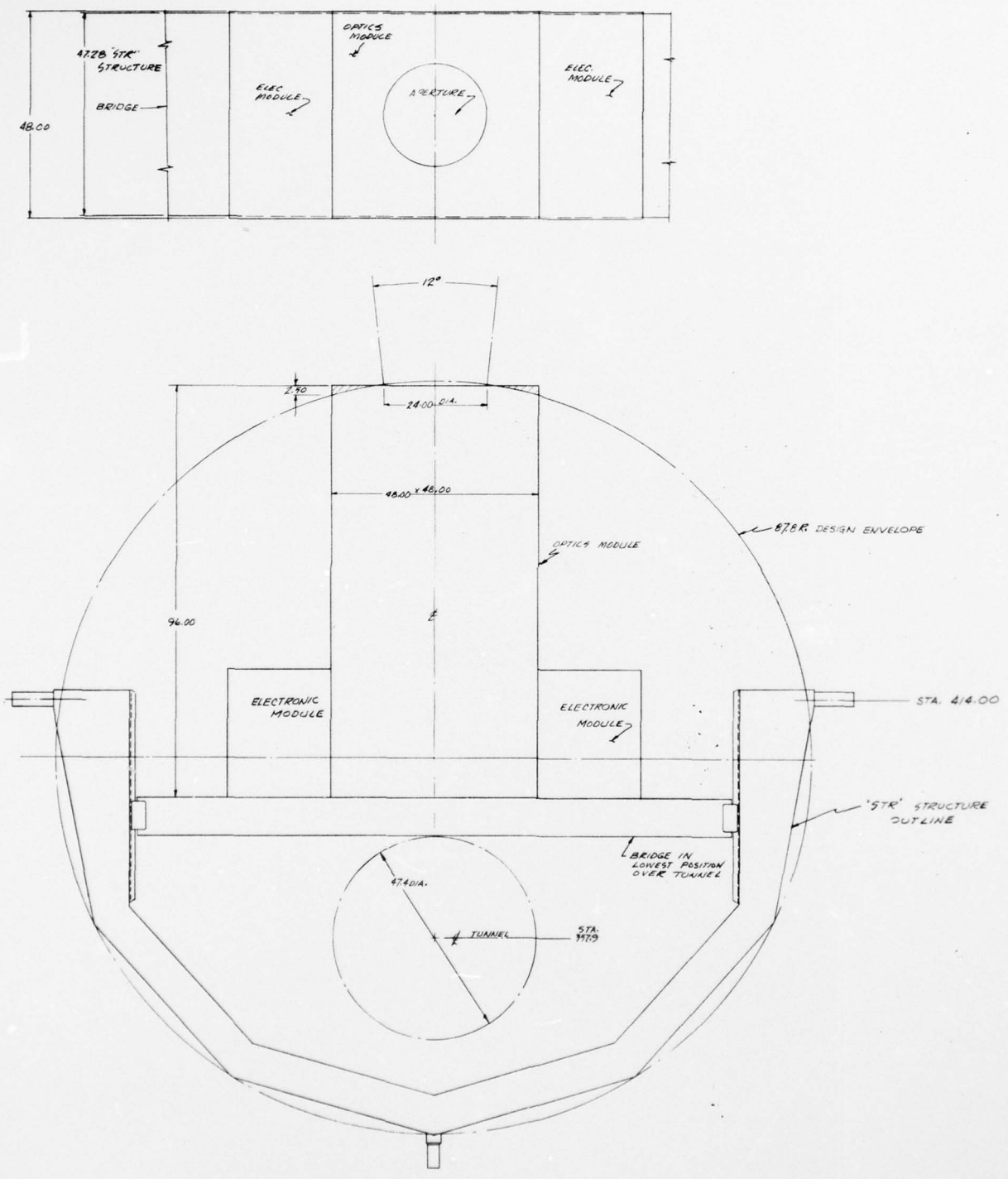




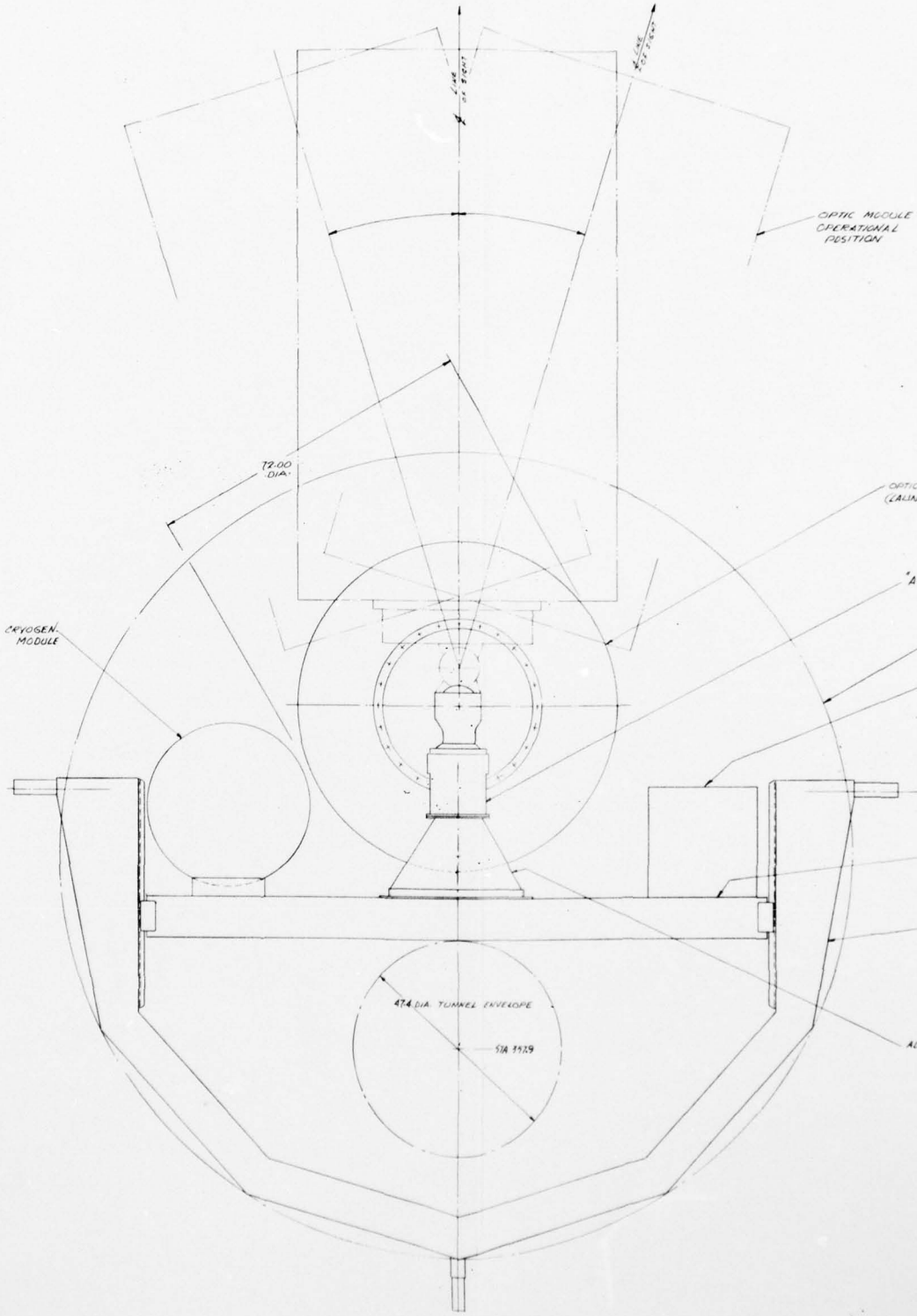
CHANGE OF PHASE
MODULE OPTION

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON	SIGNATURE	DATE	GENERAL ELECTRIC SPACE SYSTEMS DEPT. LOC. PHILA. PA.		
2 PLACE DECIMALS ±	ALL SURF	DRAWN	10/16/77	COMPONENT PANEL CONFIGURATIONS	
3 PLACE DECIMALS ±		CHECKED			
ANGLES ±		ENGR			
FRACTIONS ±	✓	MFG			
MATERIAL		QUALITY			
				SIZE J	CODE IDENT NO. STR-003
				SCALE 1/4	SHEET 3 OF 3

2/



#1 LASERCOM RECEIVER TEST

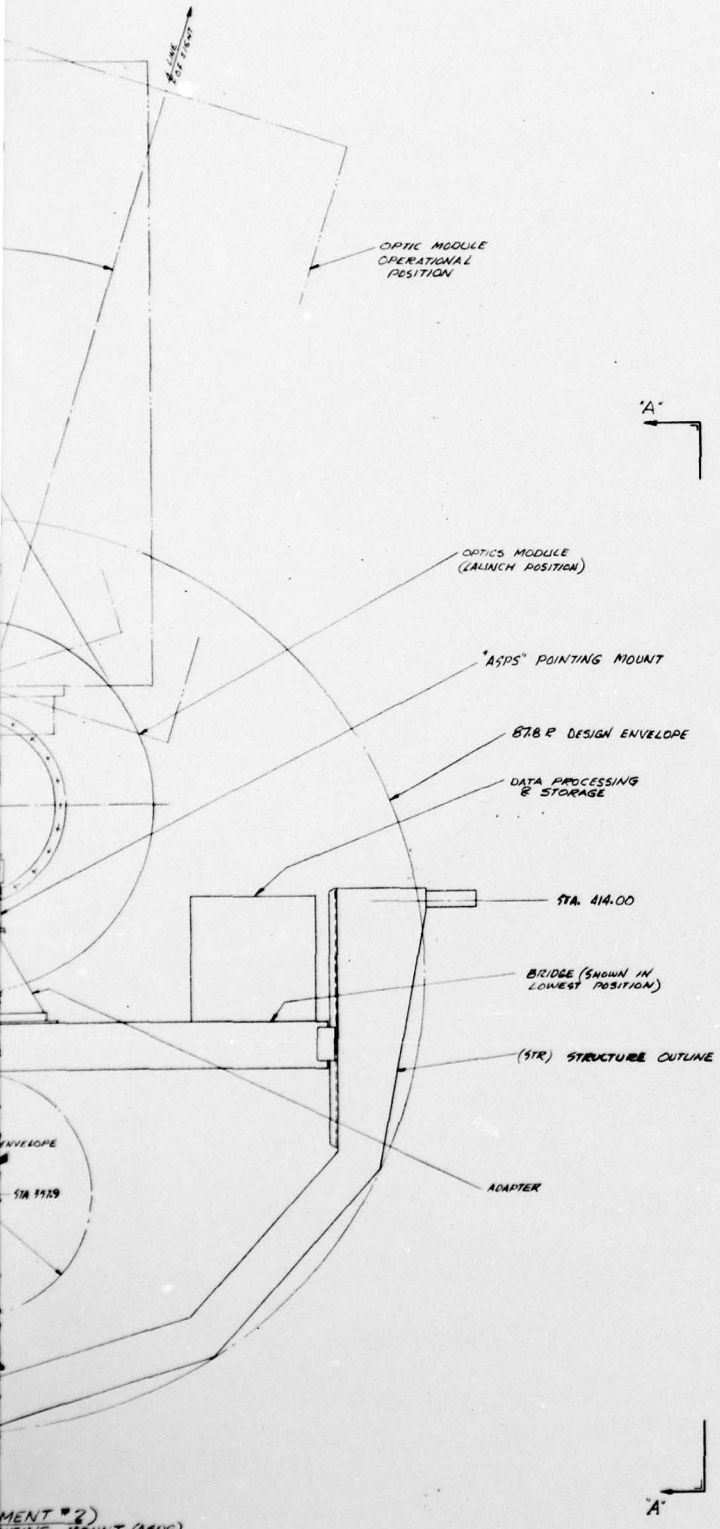


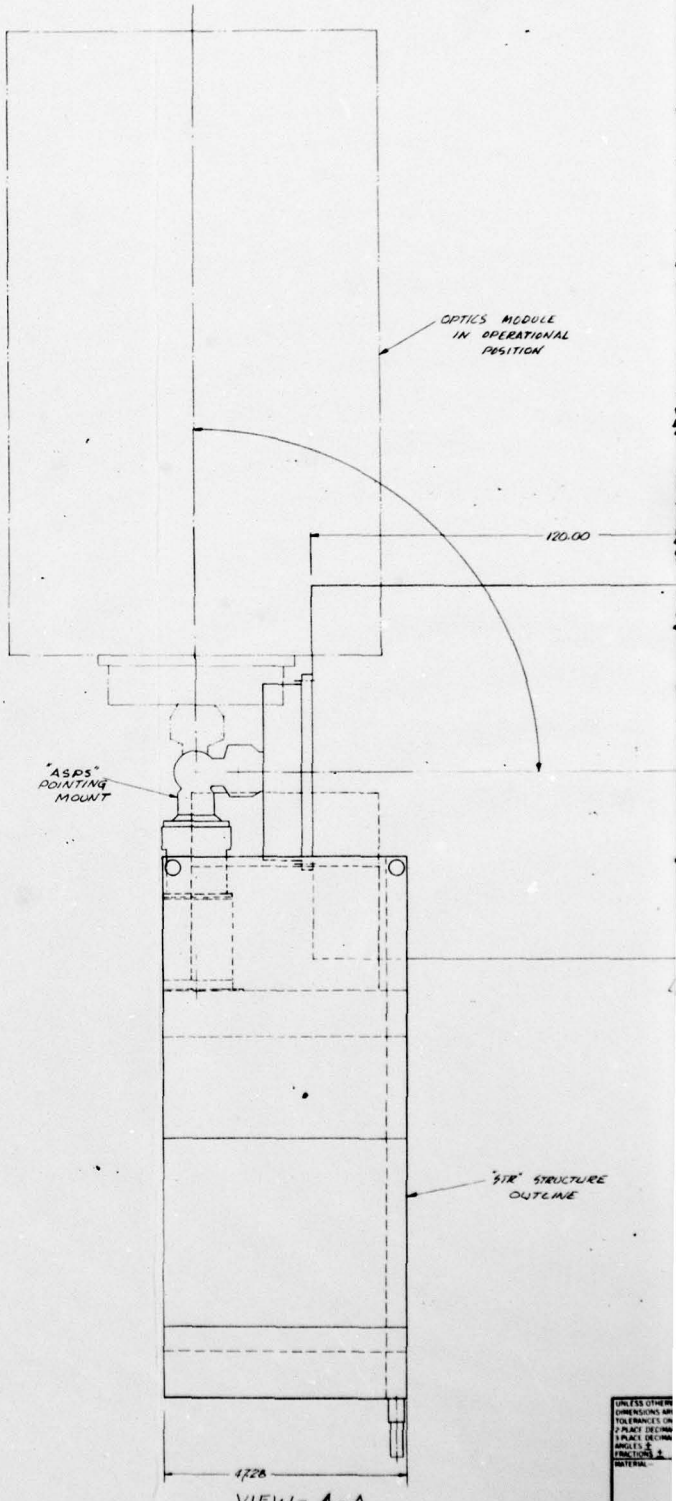
HIRISE (EXPERIMENT #2)
MOUNTED ON A POINTING MOUNT (ASPS)
SHOWN ON (STR) STRUCTURE.
SCALE 1/20

VELOPE

STA. 414.00

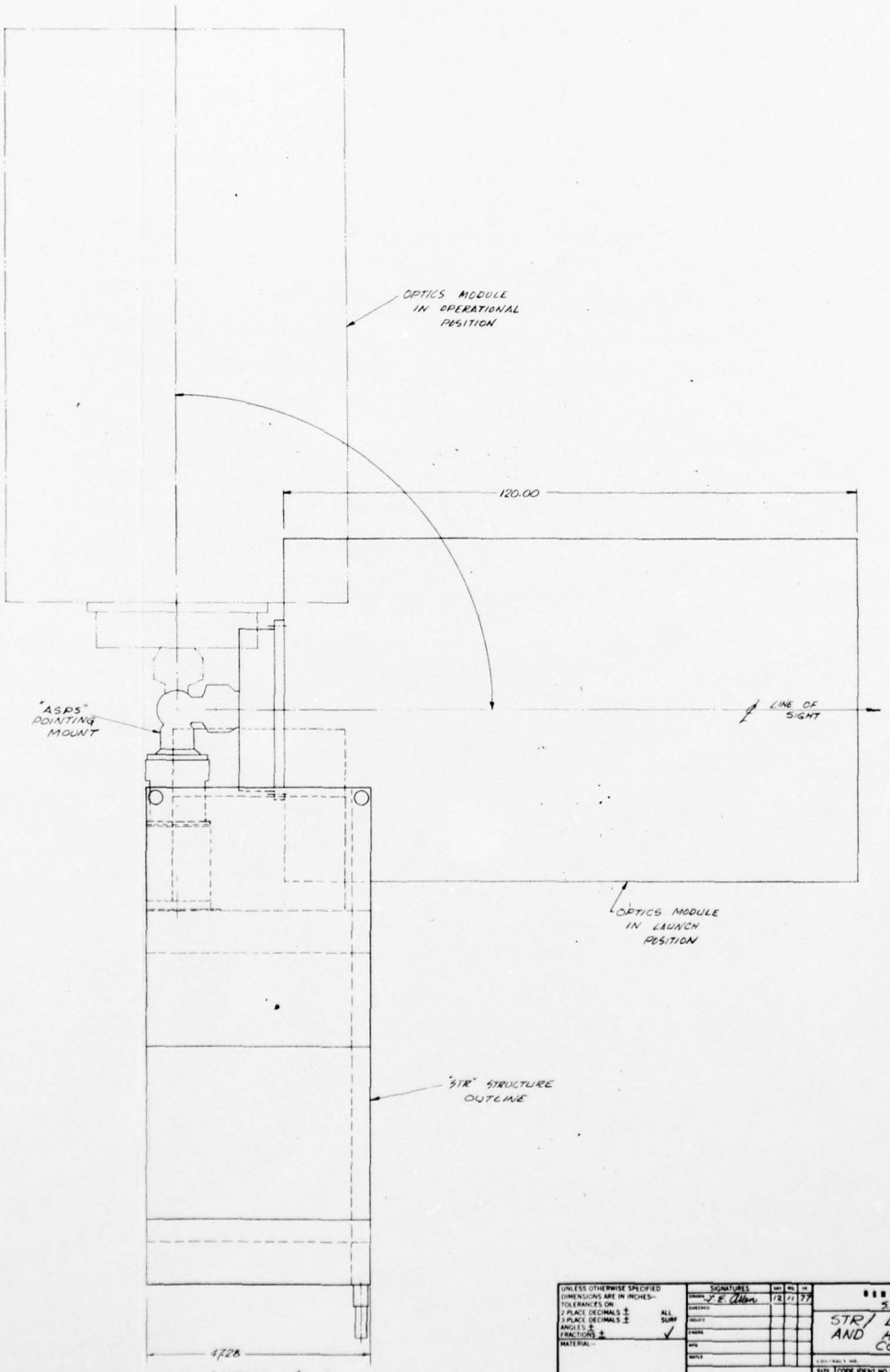
TR' STRUCTURE OUTLINE





VIEW - A - A
SCALE 1/10

MENT "2"
TING MOUNT (ASPS)
TRUCTURE.



VIEW - A - A
SCALE 1/10

"A"

"A"

...G MOUNT

...IGN ENVELOPE

...ESSING ...GE

414.00

...KNOWN IN ...POSITION)

...UCTURE OUTLINE

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON 2 PLACE DECIMALS ± 3 PLACE DECIMALS ± ANGLES ± FRACTIONS ±	ALL SURF ✓
MATERIAL —	

SIGNATURES			
DRAWN	J. E. Allen	12	11 77
CHECKED			
ISSUED			
ENGRG			
MFG			
QUALITY			

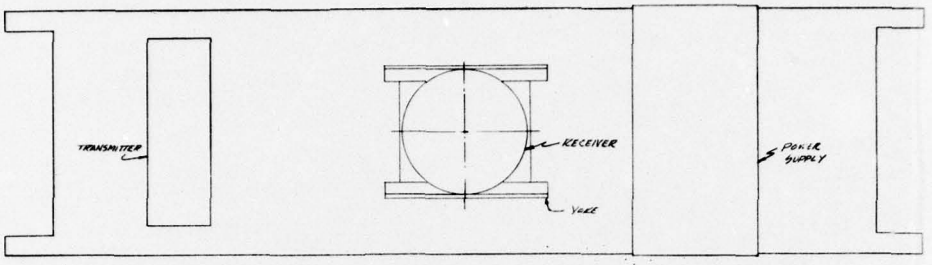
GENERAL ELECTRIC
ORD DEPT LOC PHILA., PA.

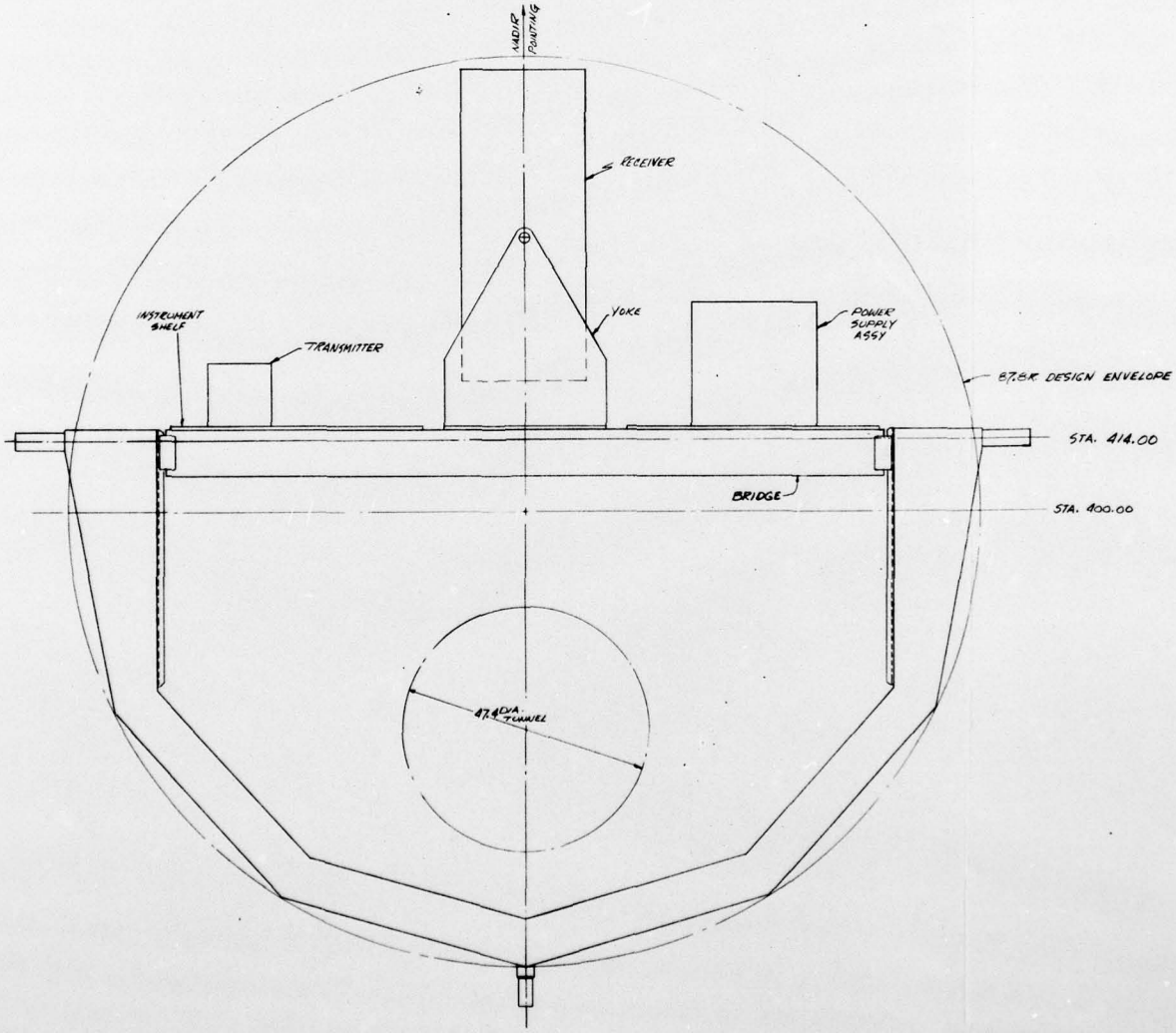
STR/ LASER RECEIVER(#1) AND HIRISE(#2) CONFIGURATION

CONTRACT NO.

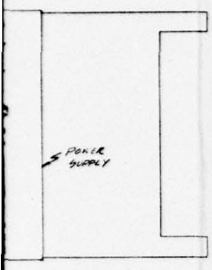
SIZE	CODE IDENT NO	STR - 004
SCALE 1/10		SHEET 1 OF 1

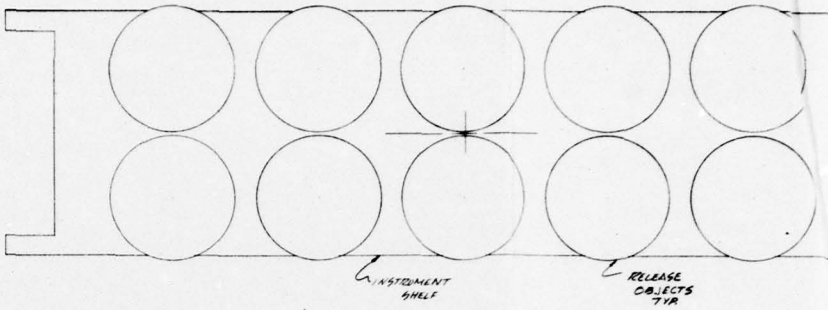
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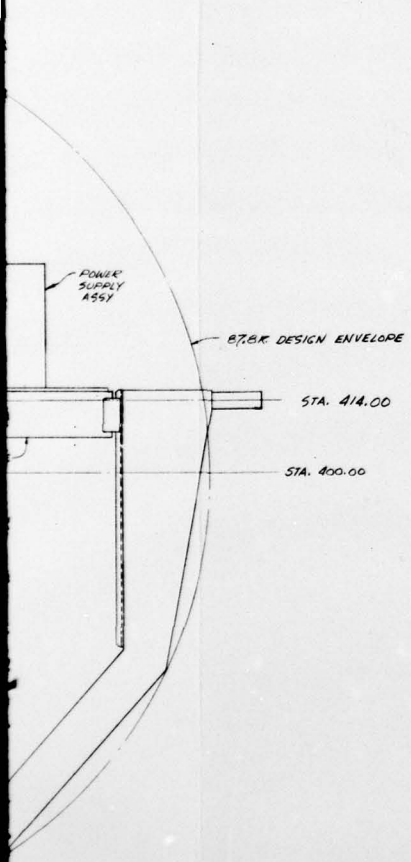


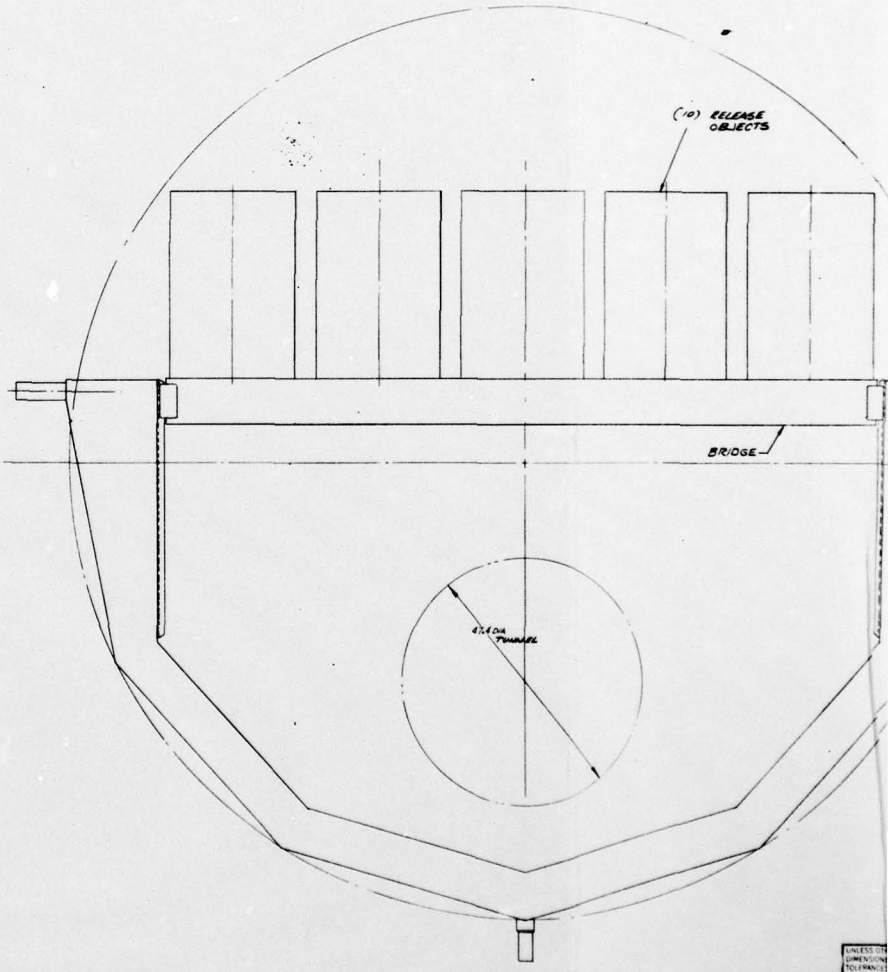


#3 ATMOSPHERIC TOPSIDE LASER SOUNDER
(ATLAS)



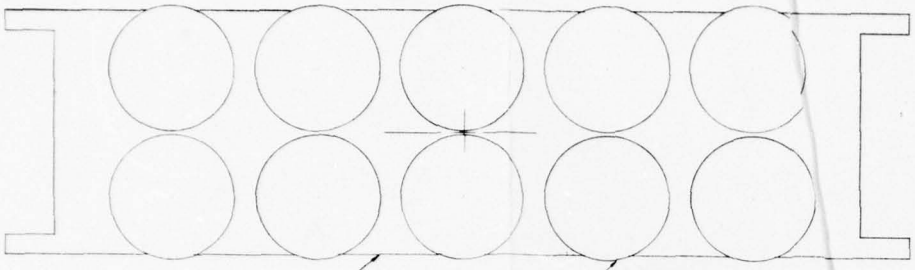


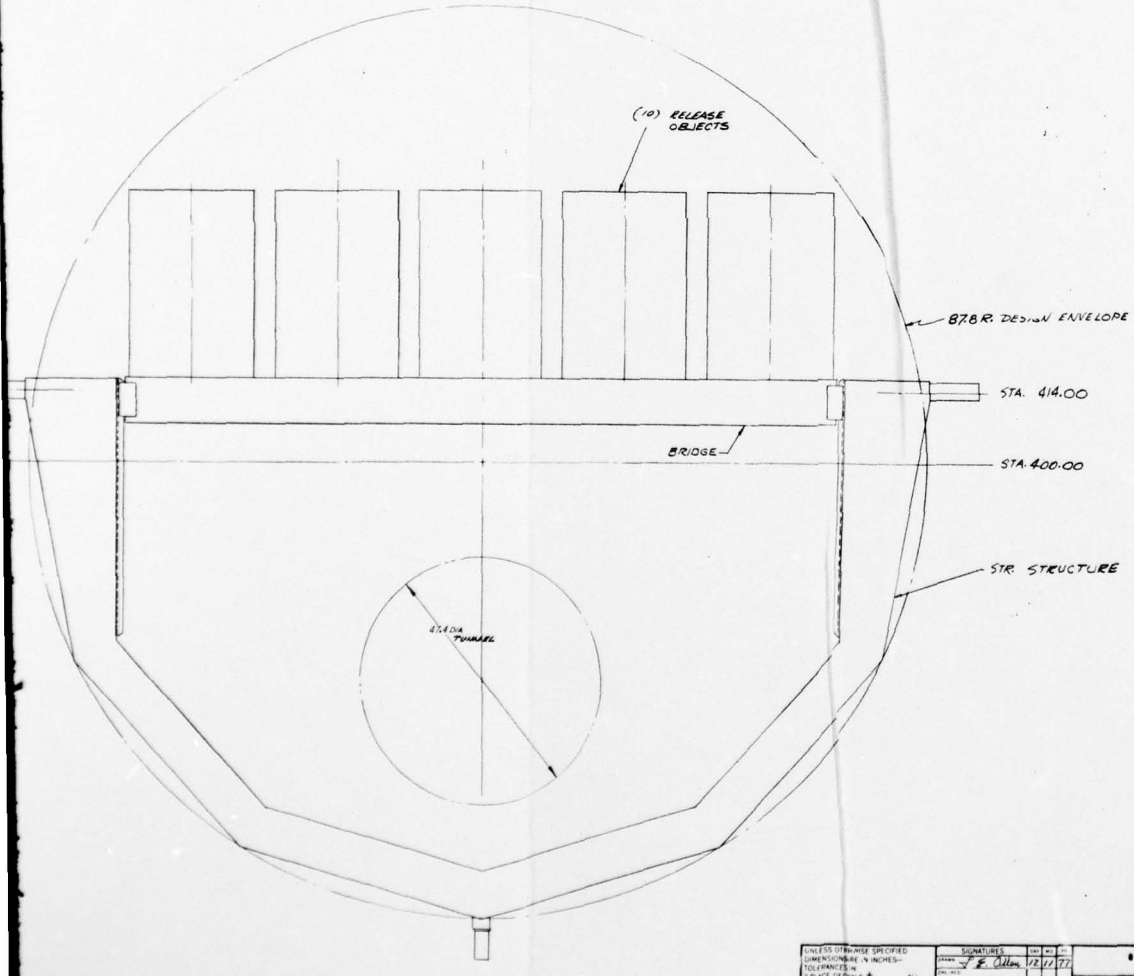




#4. PRECISION RELEASE ACCURACY TEST

2





#4. PRECISION RELEASE ACCURACY TEST

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES — TOLERANCES IN 2 PLACE DECIMALS ± 1 PLACE DECIMALS ± ANGLES ± FRACTIONS ± MATERIAL	ALL SURF ✓

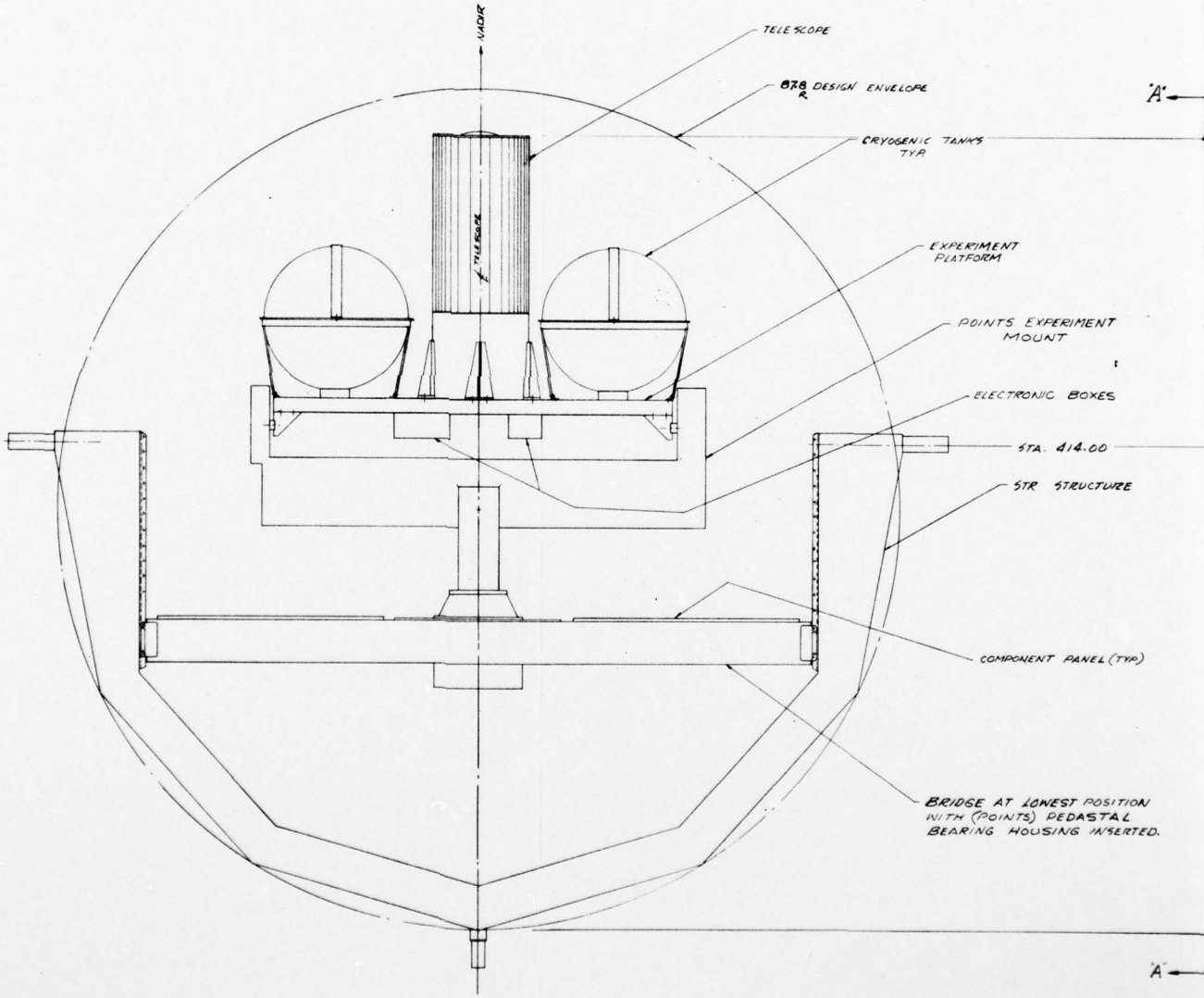
SIGNATURES			
DRAWN	J. E. Allen	12/11	77
CHK. REQ.			
DESIGN			
ENGRG			
MFG			
QA			
MATLS			

GENERAL ELECTRIC
S.S. DEPT. LOC. PHILA, PA.

STR/LASER SOUNDER(#3) AND ACCURACY RELEASE TEST(#4) CONFIGURATIONS

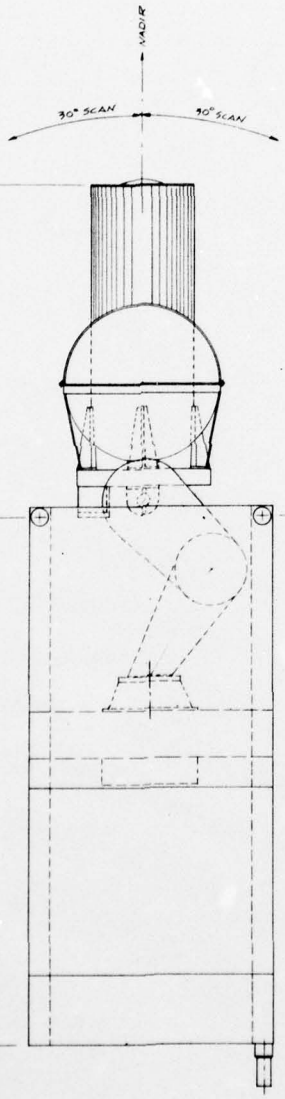
SIZE	CODE IDENT NO.	STR-005
SCALE 1/10		SHEET 1 OF 1

3



EARTH LIMB MEASUREMENT SENSOR (EXPERIMENT #5) MOUNTED ON A POINTING MOUNT (POINTS) SHOWN ON (STR) STRUCTURE

SCALE = 1/10



VIEW "A-A"
SCALE- 1/10

S EXPERIMENT
MOUNT

TRONIC BOXES

STA 414.00

STR STRUCTURE

PONENT PANEL (TYP)

LOWEST POSITION
TS) PEDASTAL
HOUSING INSERTED.

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES— TOLERANCES ON		SIGNATURES	DAY	MO	YR
2 PLACE DECIMALS ±	ALL	DRAWN J.E. Allen	12	10	77
3 PLACE DECIMALS ±	SURF	CHECKED			
ANGLES ±	✓	STRESS			
FINISH ±		FINISH			
MATERIAL—		APPD			
		DATE			

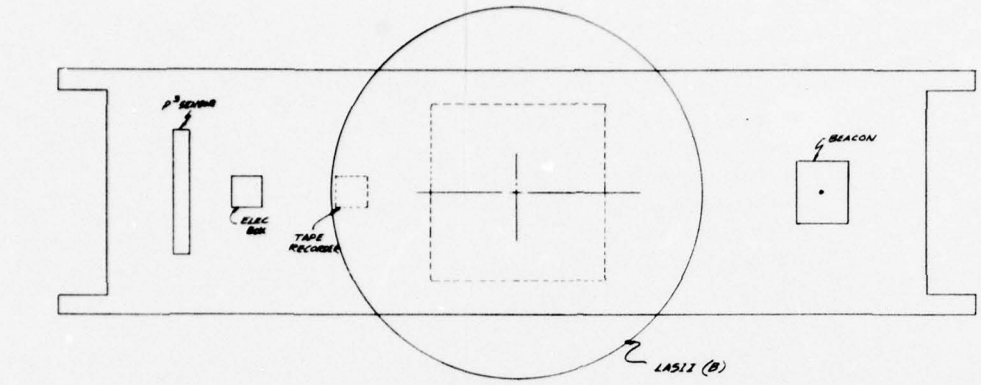
GENERAL ELECTRIC
SS DEPT. VALLEY FORGE, PA.

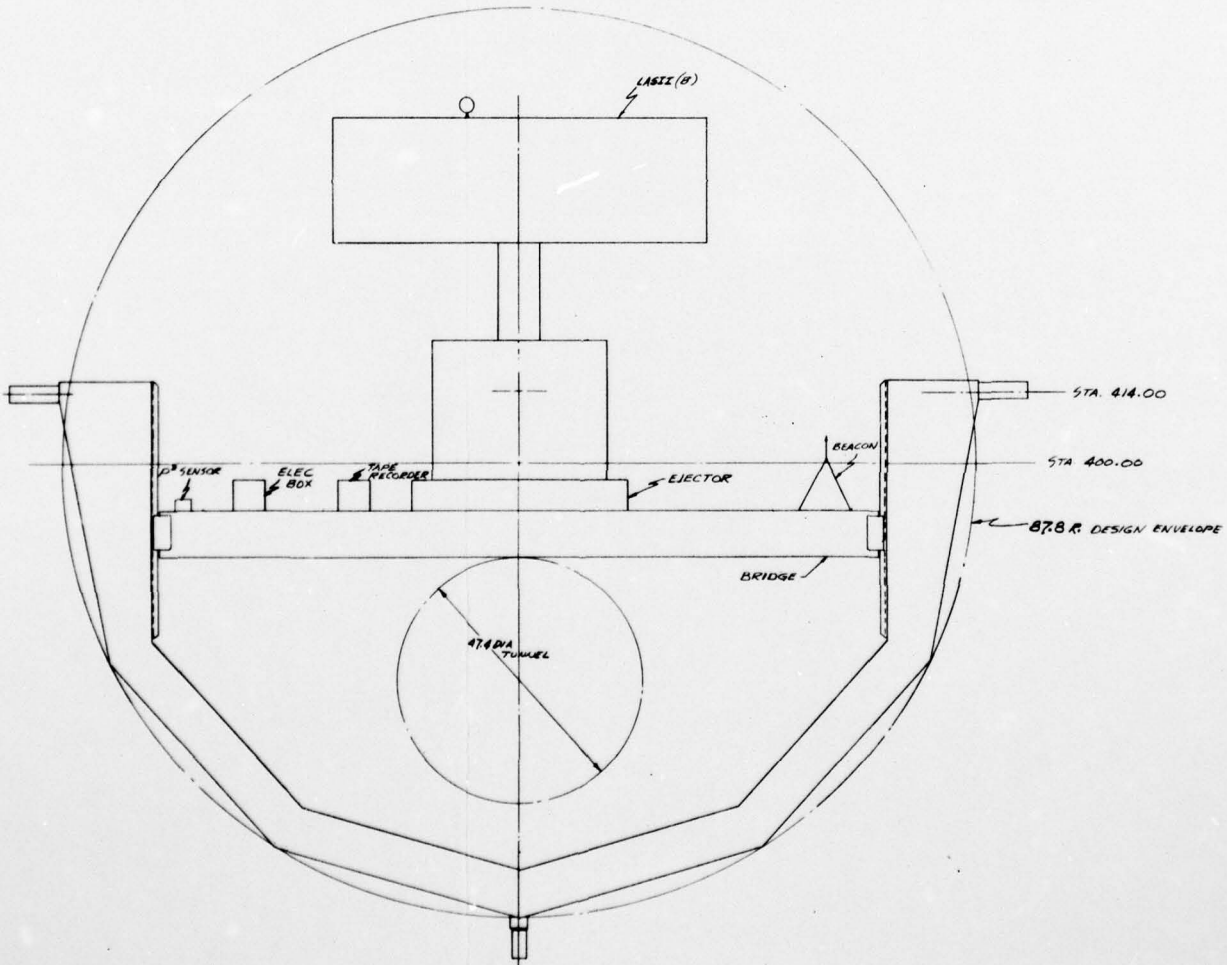
STR/EARTH LIMB MEASUREMENT SENSOR (#2) CONFIGURATION

CONTRACT NO.

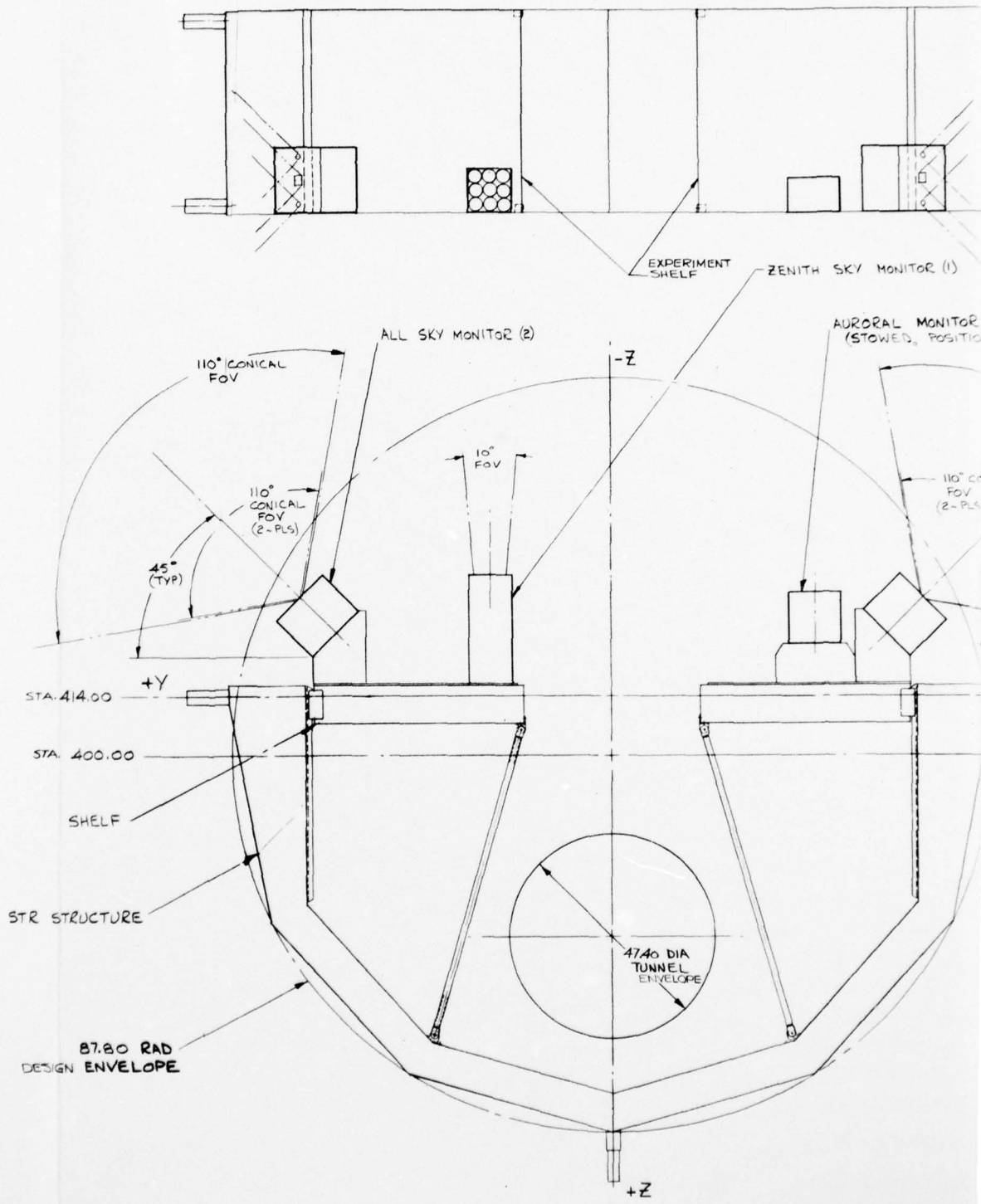
SIZE	CODE IDENT NO	STR-006
SCALE 1/10		SHEET 1 of 1

2





6 LOW ALTITUDE STUDY OF IONOSPHERIC IRREGULARITIES (LASS II)



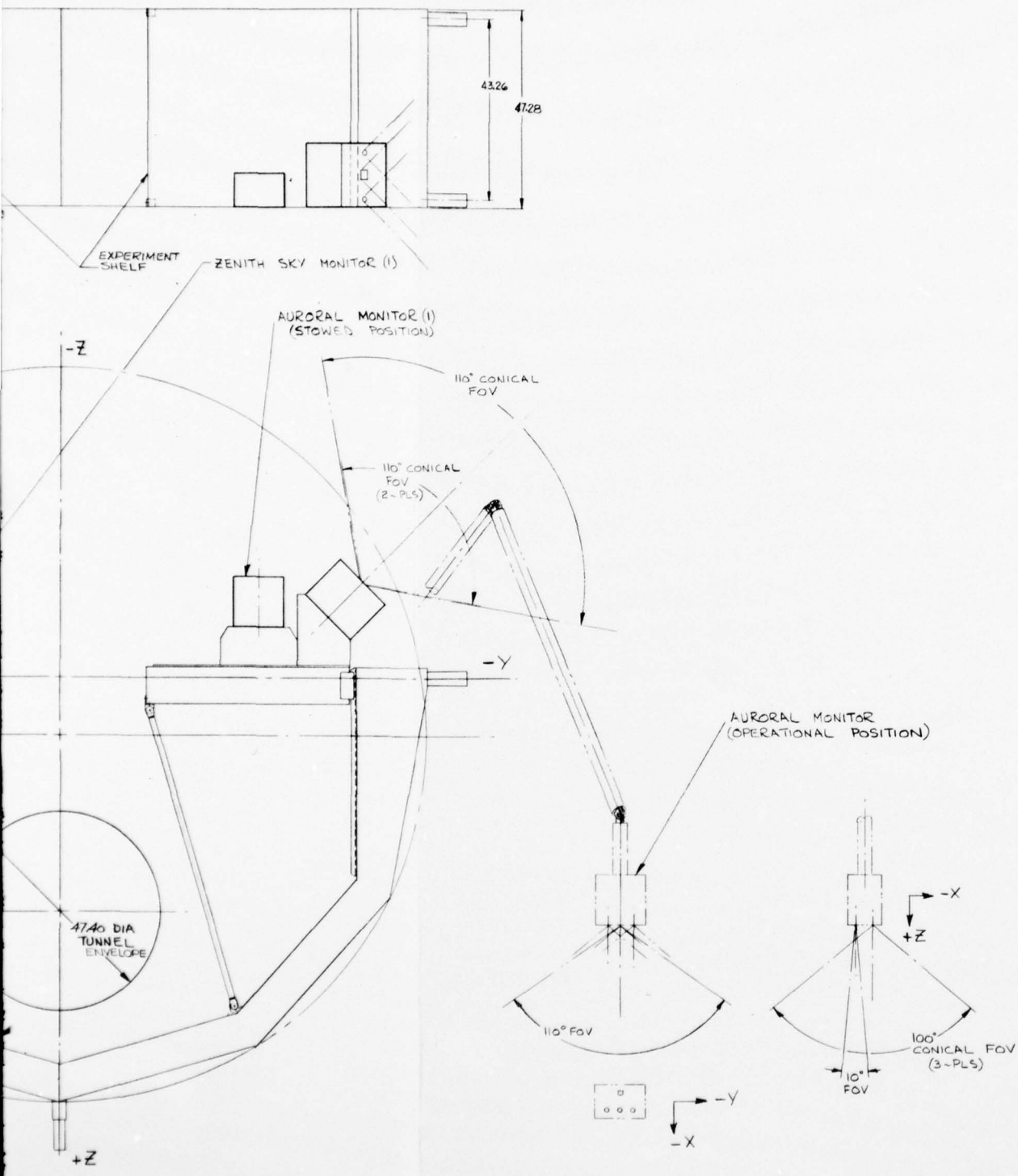
VACUUM

4 S

STA. 414.00

STA. 400.00

R. DESIGN ENVELOPE



VACUUM ULTRAVIOLET PHOTOMETER & SPECTROMETER

7

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES ON
2 PLACE DECIMALS ± ALL SURF
3 PLACE DECIMALS ±
ANGLES ± ✓
FRACTIONS ±
MATERIAL –

SIGNATURES — DRAWN J. E. Wm 12 11 77

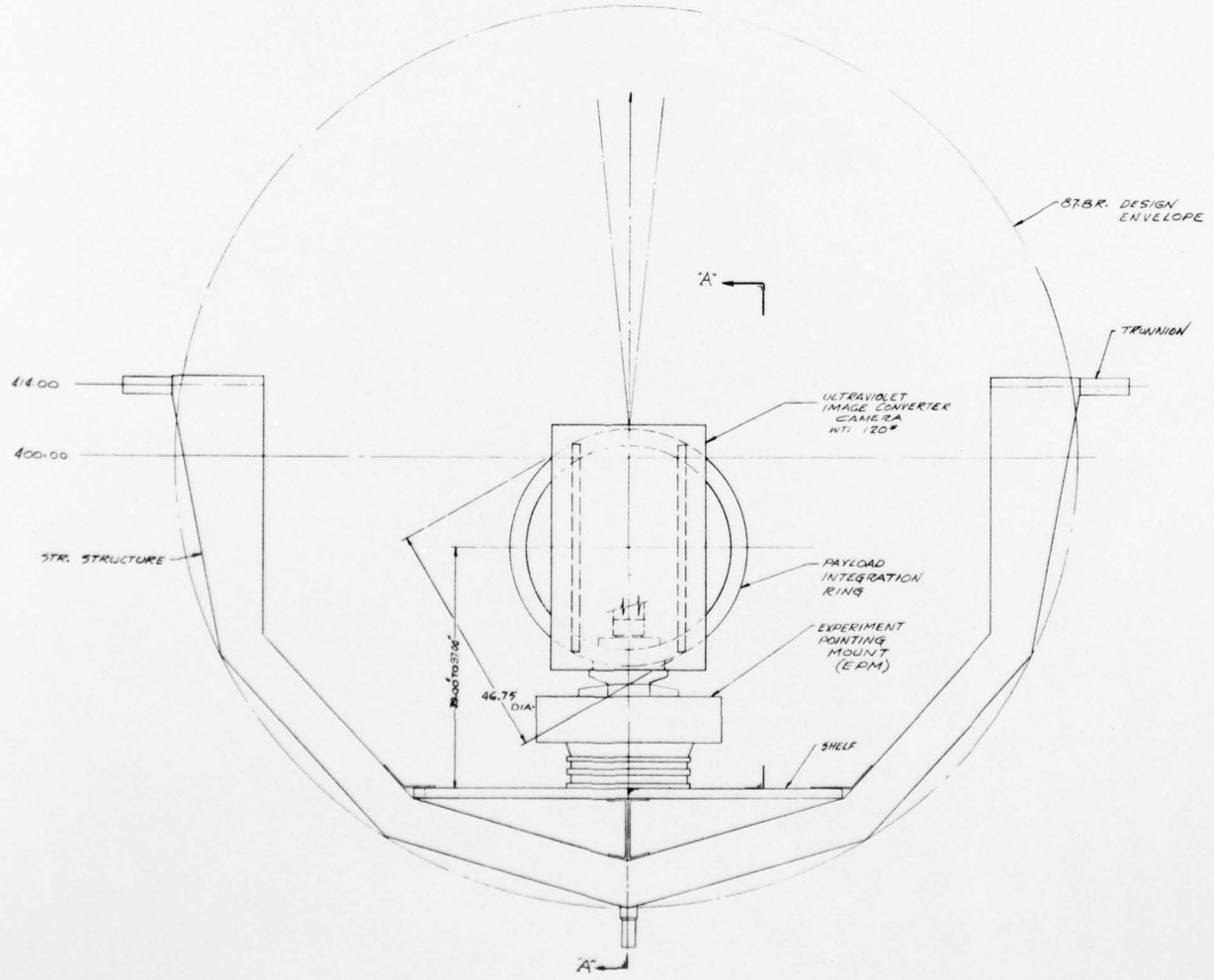
GENERAL ELECTRIC S.S. DEPT. LOC. PHILA, PA.

STR/IONOSPHERIC IRREGULARITIES (#6) AND POTOMETER AND SPECTROMETER (#7) CONFIGURATIONS

CONTRACT NO.

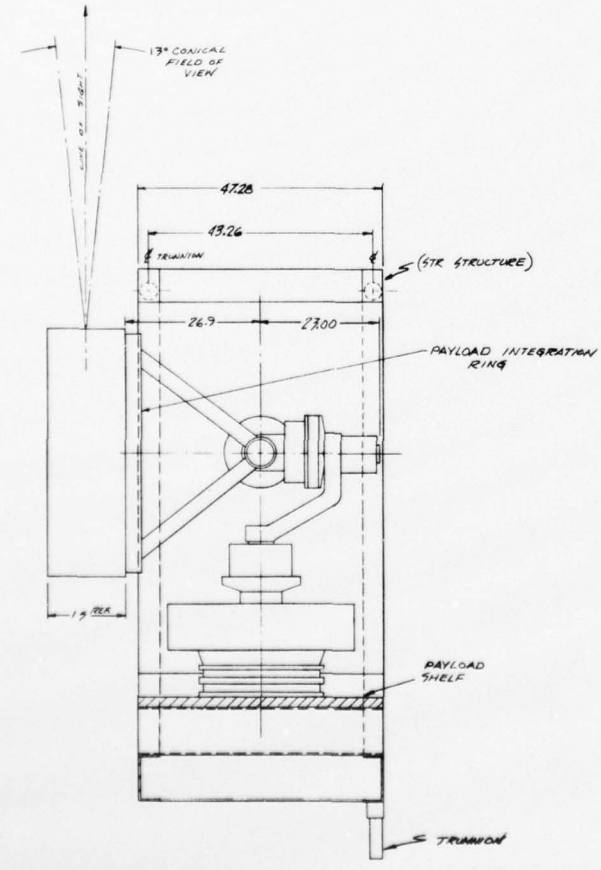
SIZE | CODE IDENT NO. | STR-007

SCALE 1/10 | SHEET 1 OF 1

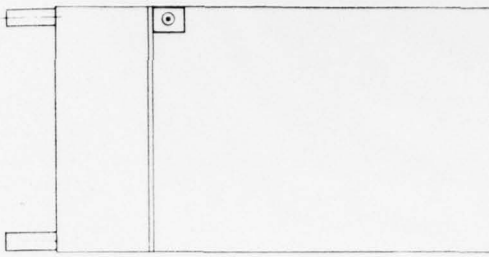


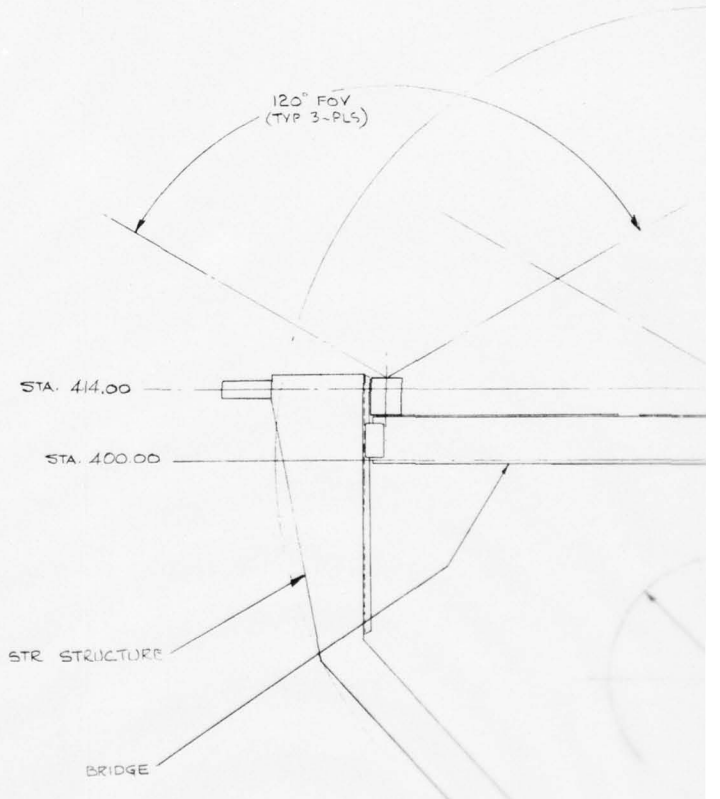
OPTICAL GEOPHYSICS AND ASTRONOMICAL OBSERVATORY (EXPERIMENT #5) MOUNTED ON AN EXPERIMENT POINTING MOUNT (IPS) SHOWN ON (STR) STRUCTURE

SCALE - 1/10



SECTION "A-A"
SCALE - 1/10





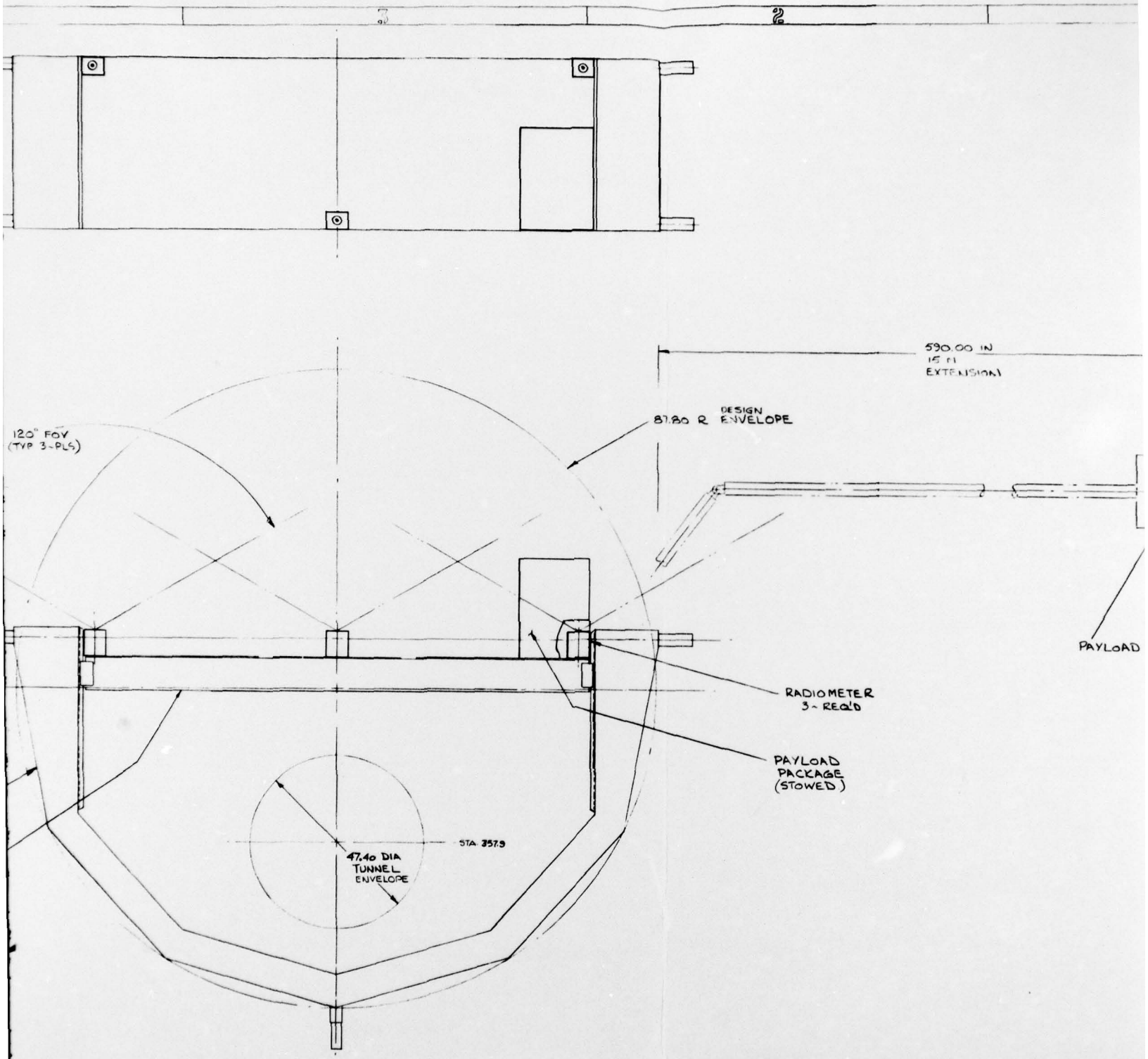
AD-A064 773 GENERAL ELECTRIC CO PHILADELPHIA PA SPACE DIV F/G 22/2
STANDARD TEST RACK EXPLOITATION STUDY.(U)
DEC 77 W P ENGLE

UNCLASSIFIED F04701-77-C-0116
SAMSO-TR-77-190 NL

5 OF 5
AD
A064773

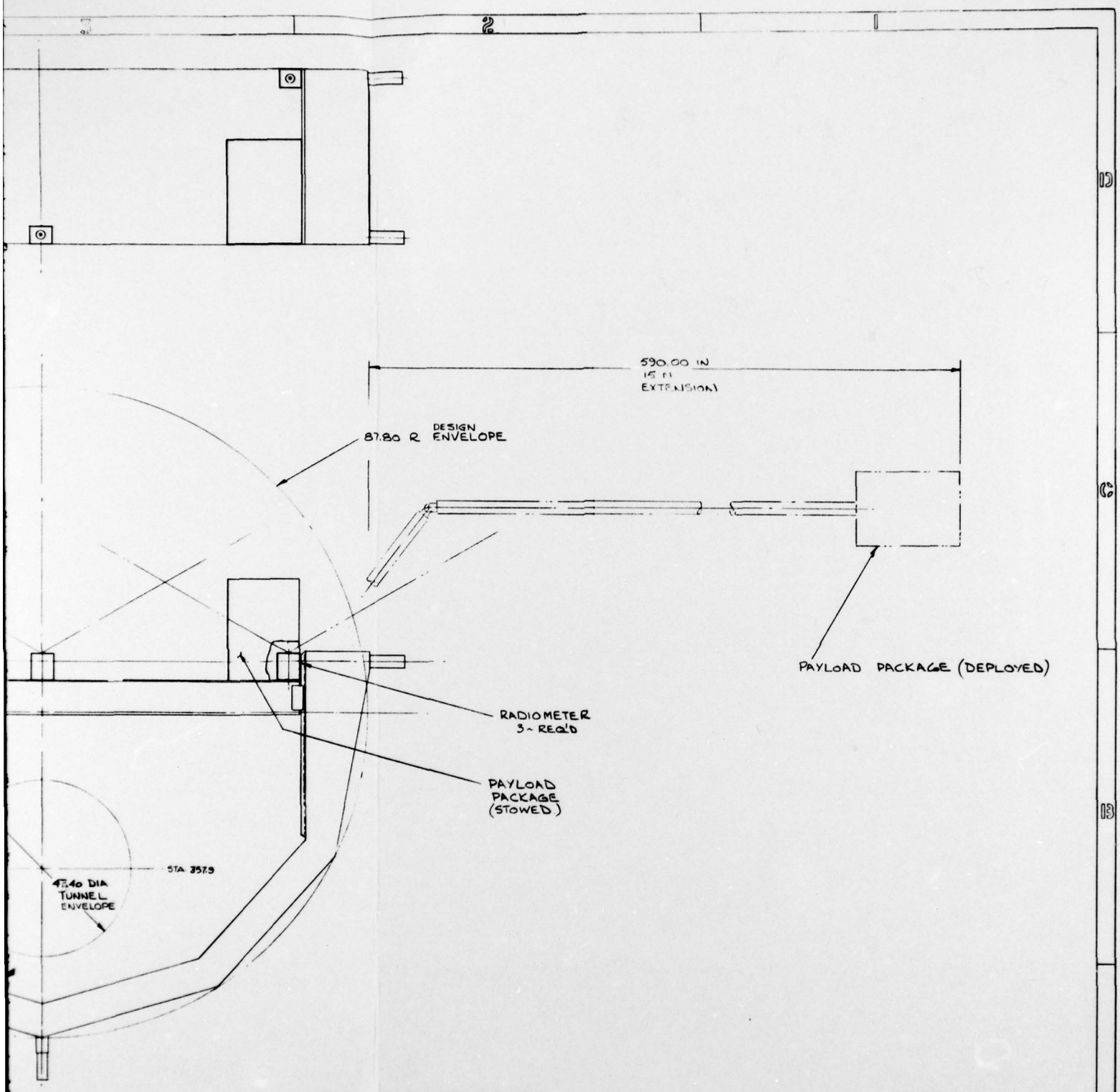


END
DATE
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OPTICAL COUNTERMEASURES
DEMONSTRATION (*11)

3

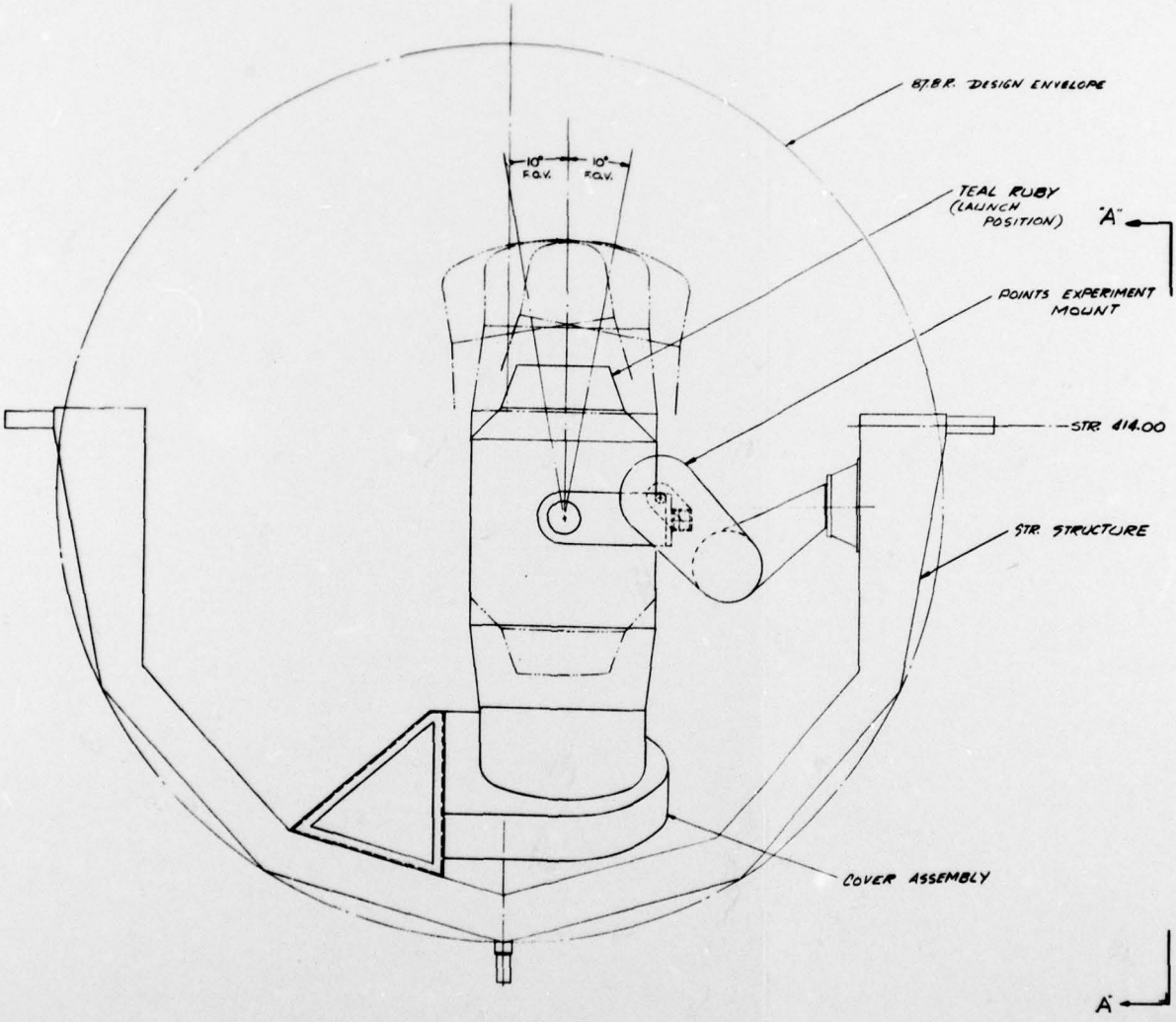


OPTICAL COUNTERMEASURES
DEMONSTRATION (*11)

GENERAL ELECTRIC
STR] OPTICAL GEOPHYSICS AND OPTICAL COUNTER-MEASURES (*11) CONFIGURATIONS
STR-008

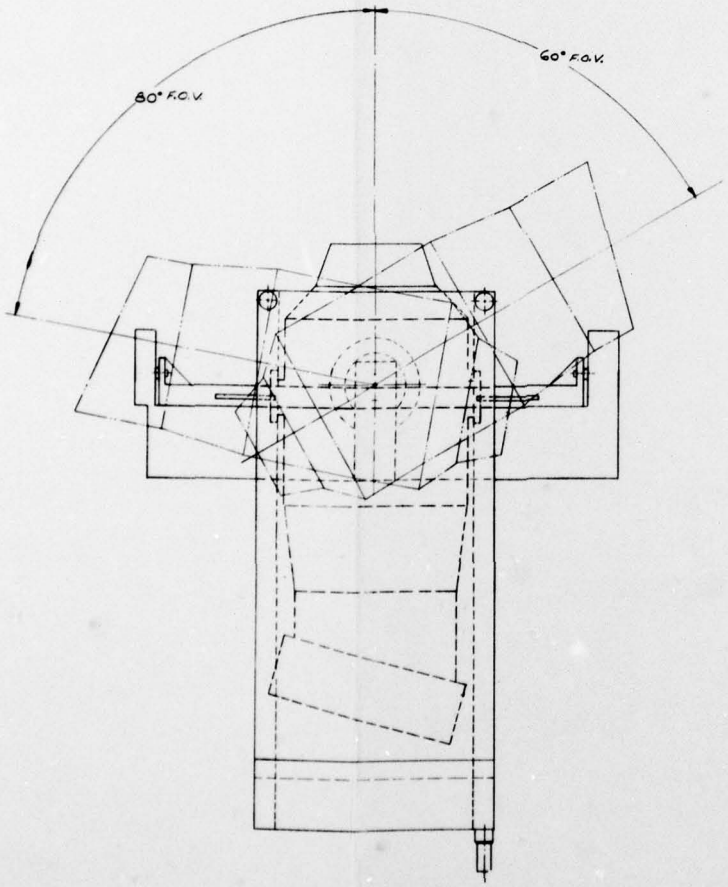
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4



TEAL RUBY (EXPERIMENT #3) MOUNTED ON A POINTING MOUNT (POINTS SHOWN ON (STR) STRUCTURE

SCALE 1/10

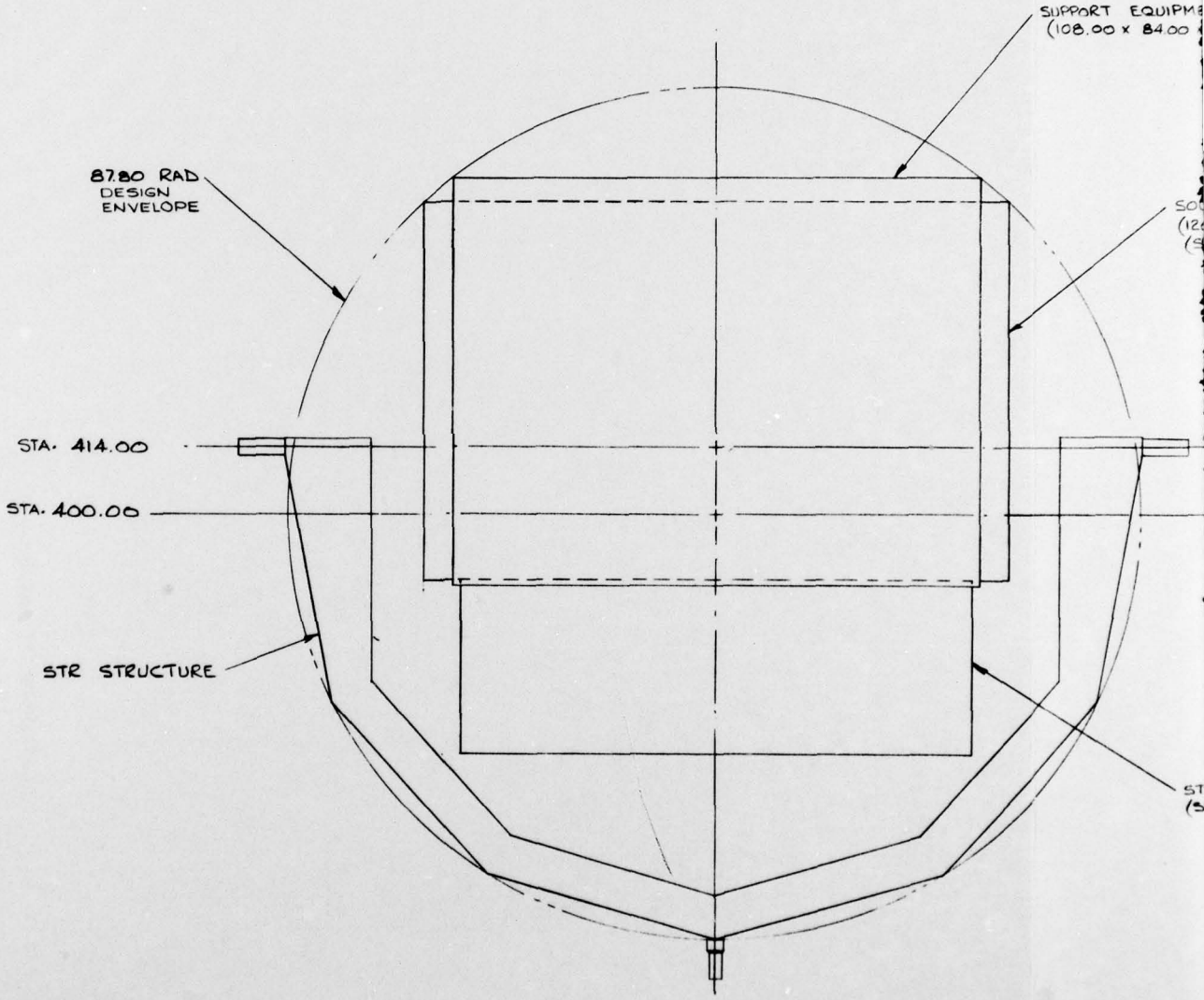


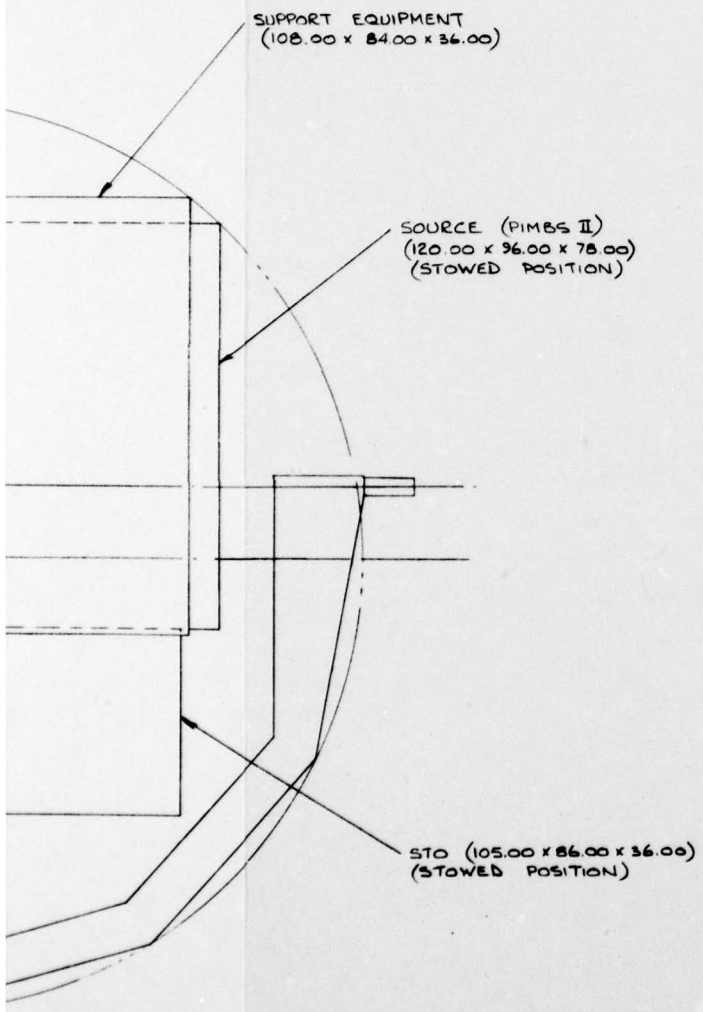
VIEW "A-A"
SCALE 1/10

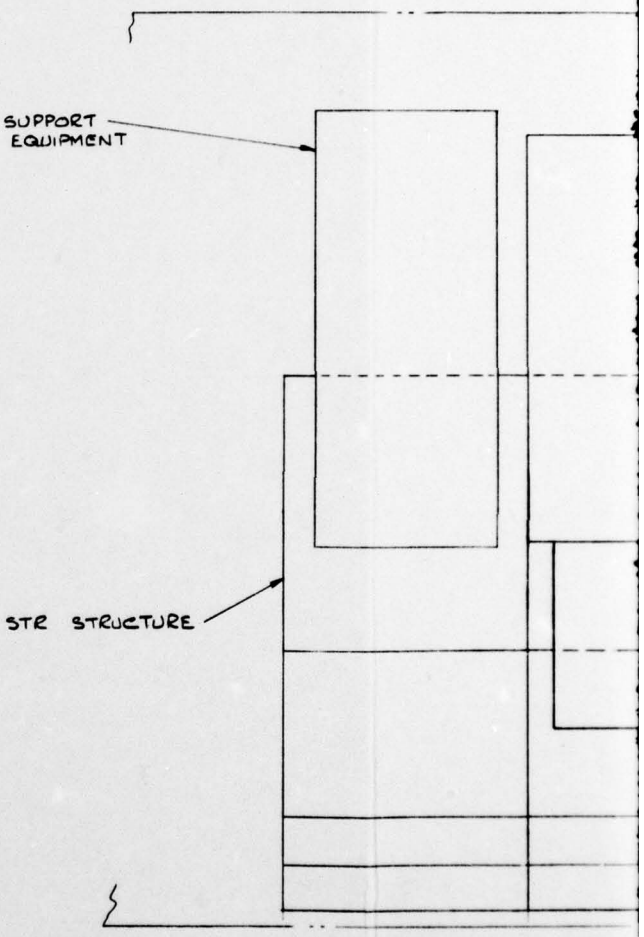
STR/ TEAL RUBY (°9) CONFIGURATION

STR-009

2


SUPPORT EQUIPME
(108.00 x 84.00
87.80 RAD
DESIGN
ENVELOPE
STA. 414.00
STA. 400.00
STR STRUCTURE

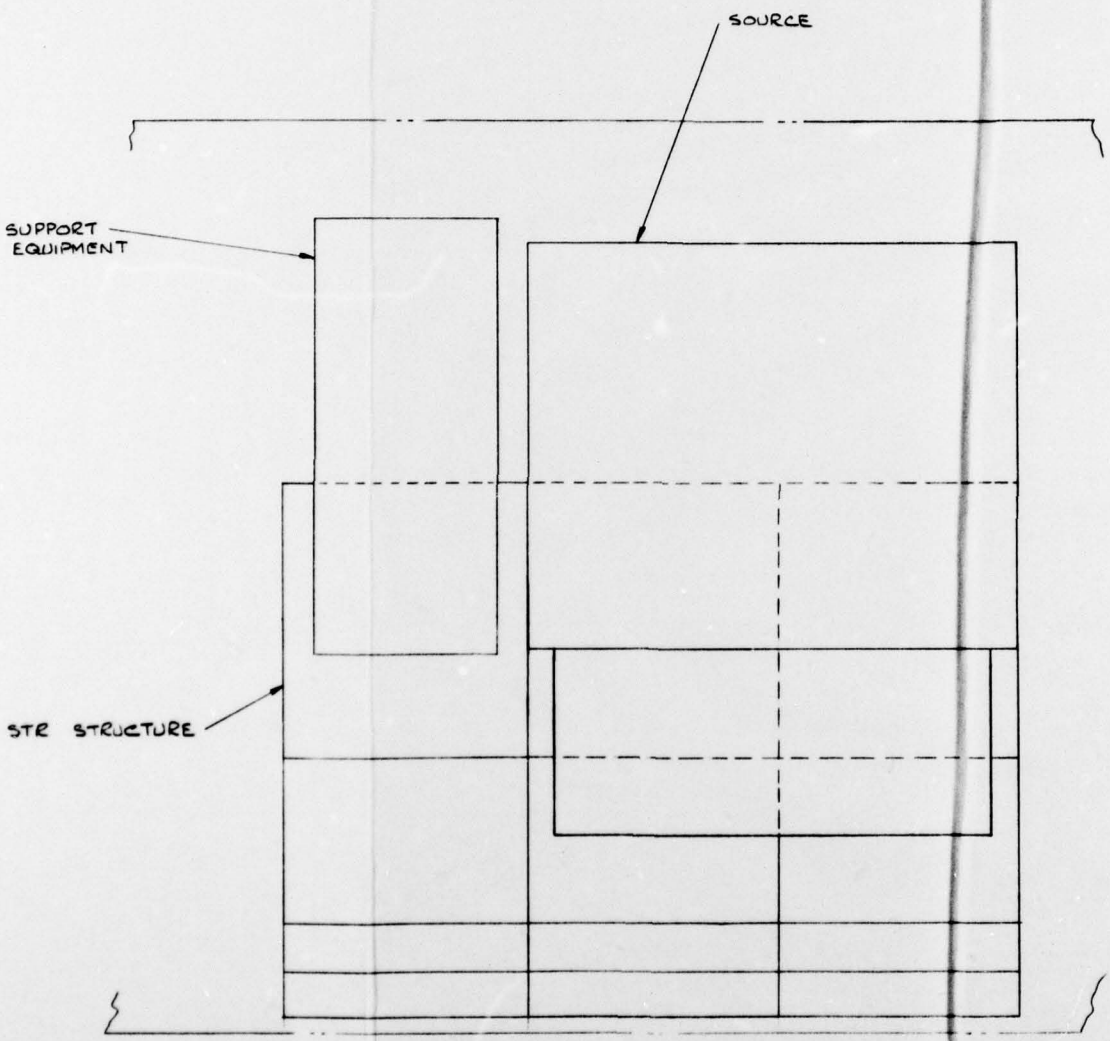




PIMBS II CONCEPT 1B

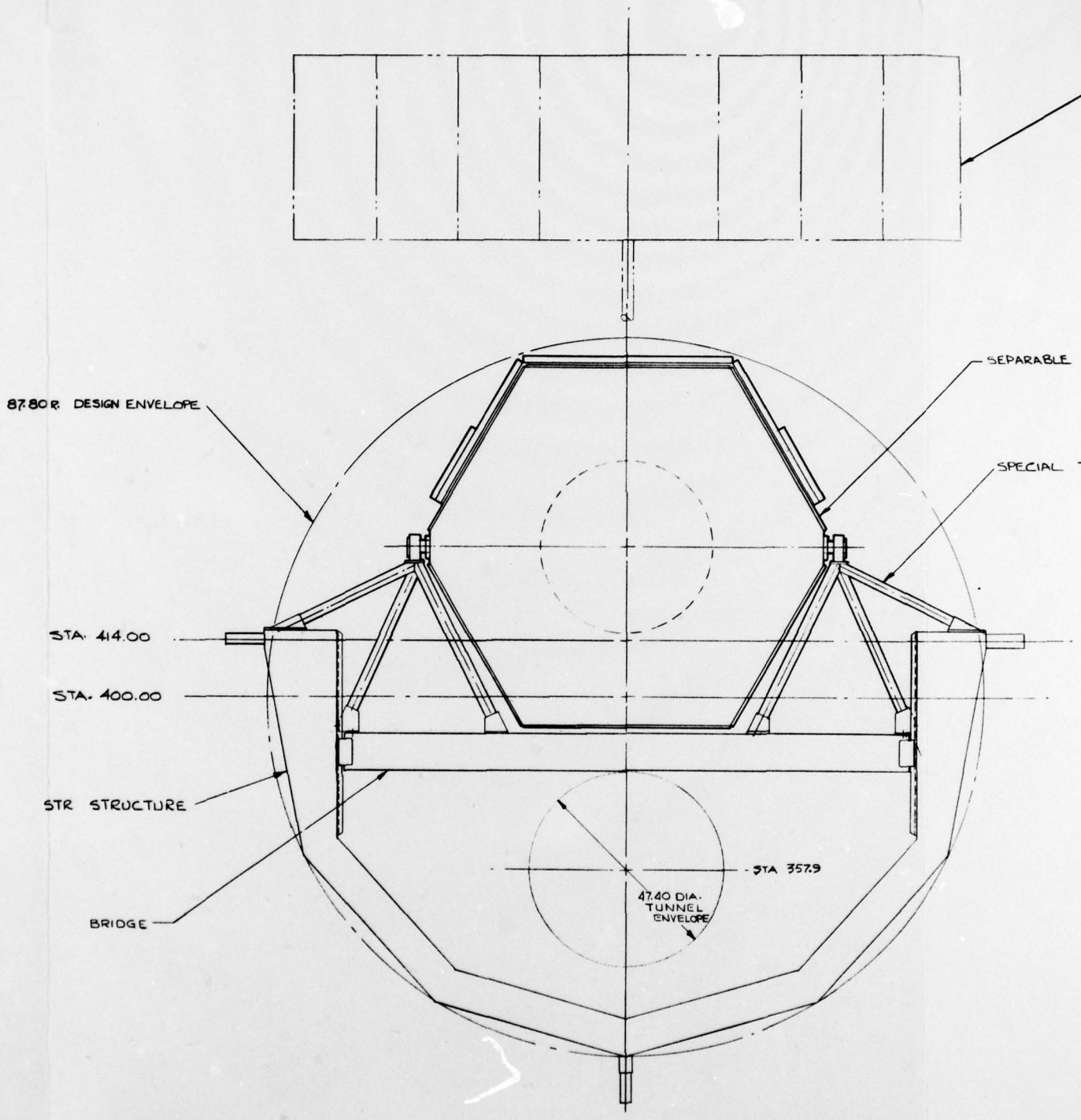
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12



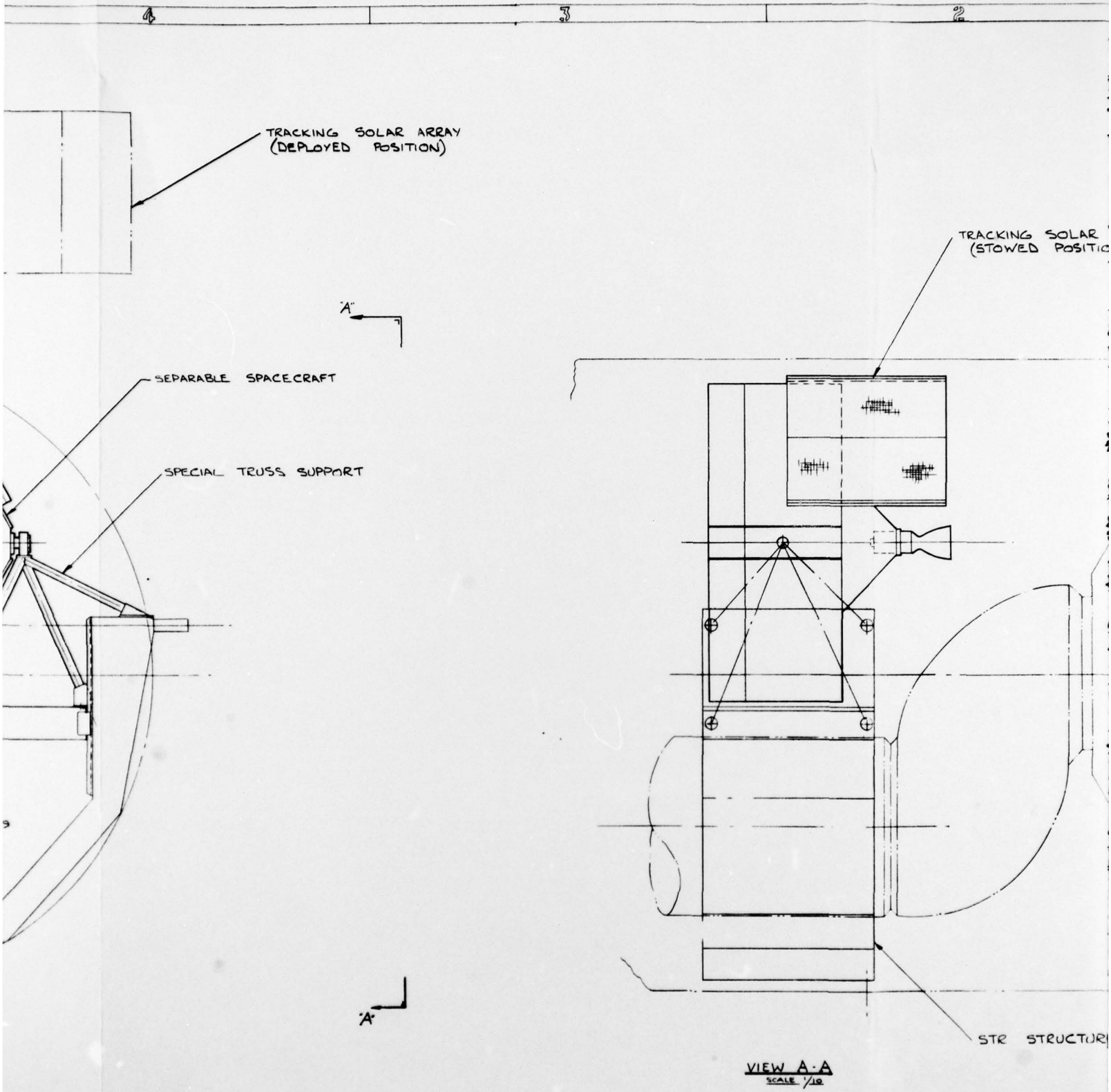
UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON 2 PLACE DECIMALS ± 3 PLACE DECIMALS ± ANGLES ± FRACTIONS ± MATERIAL —	ALL SURF ✓	SIGNATURES DRAWN CLH 12/15/77	GENERAL ELECTRIC P.S. PHILA. PA.
			STR PIMBS II CONCEPT 1B (2B) CONFIGURATION
			SIZE CODE IDENT NO. STR-010
			SCALE 1/10 SHEET 1 OF 1

3

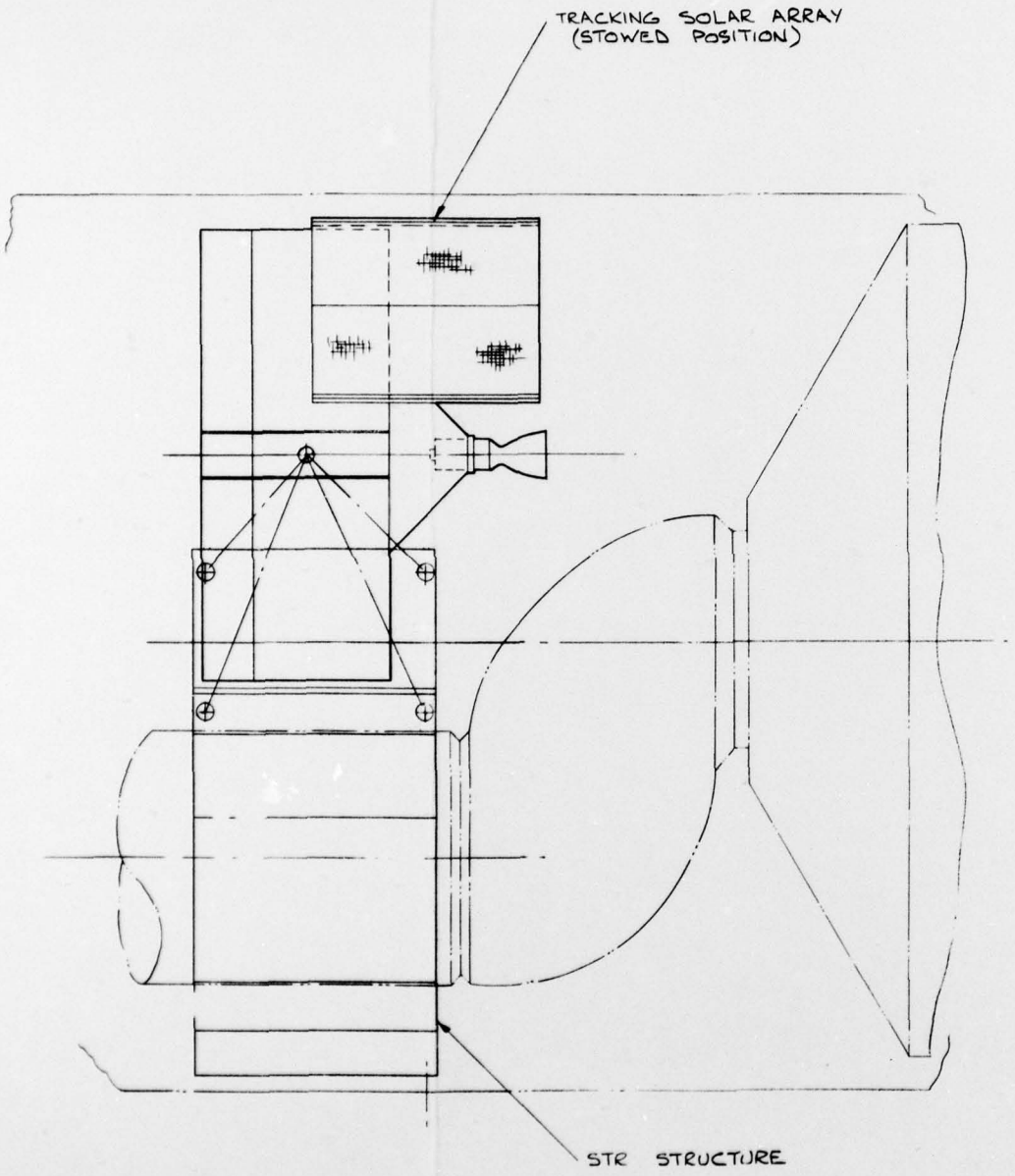


12 DEPLOYABLE SPACECRAFT

SCALE - 1/10



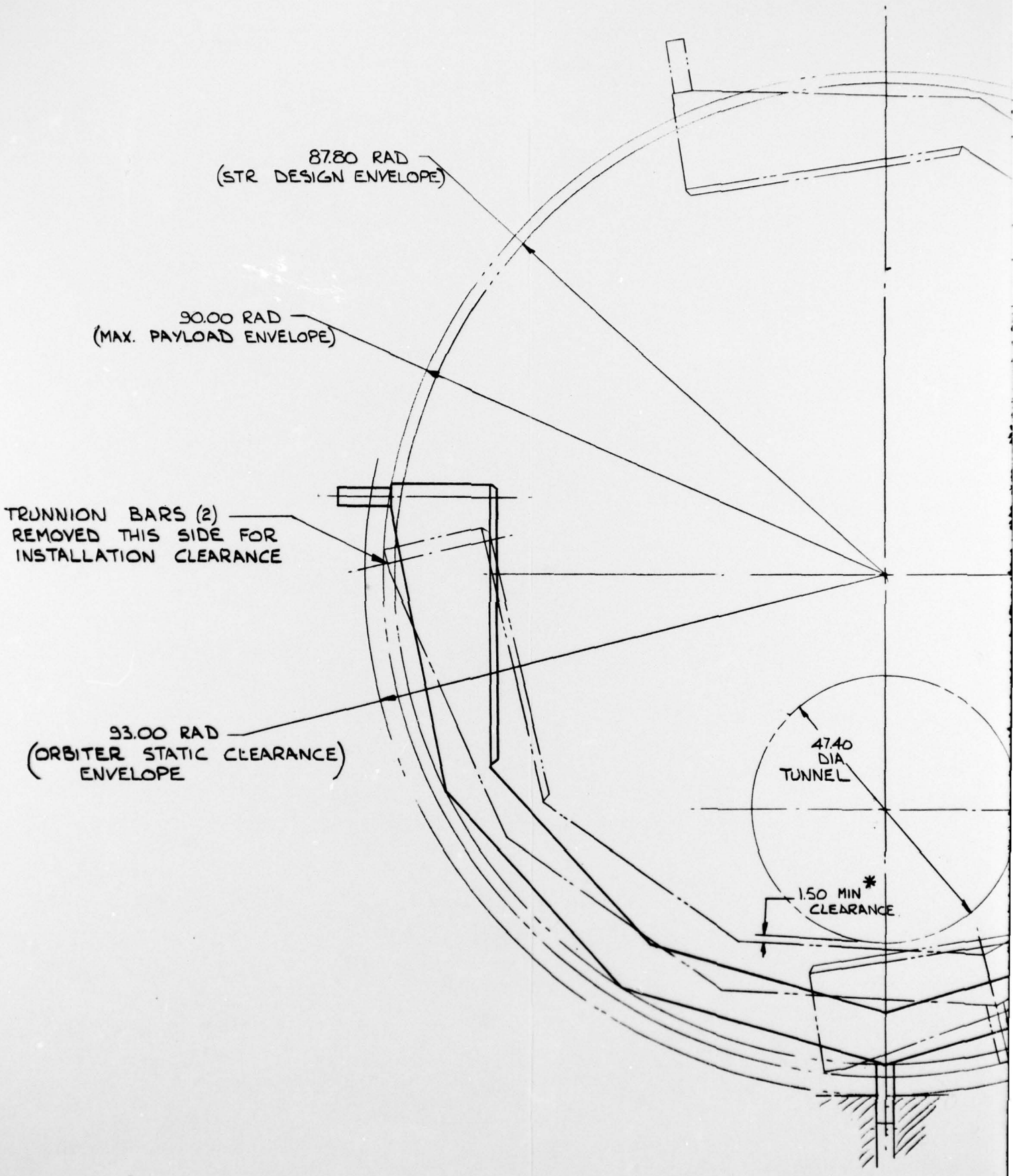
VIEW A-A
SCALE 1/10

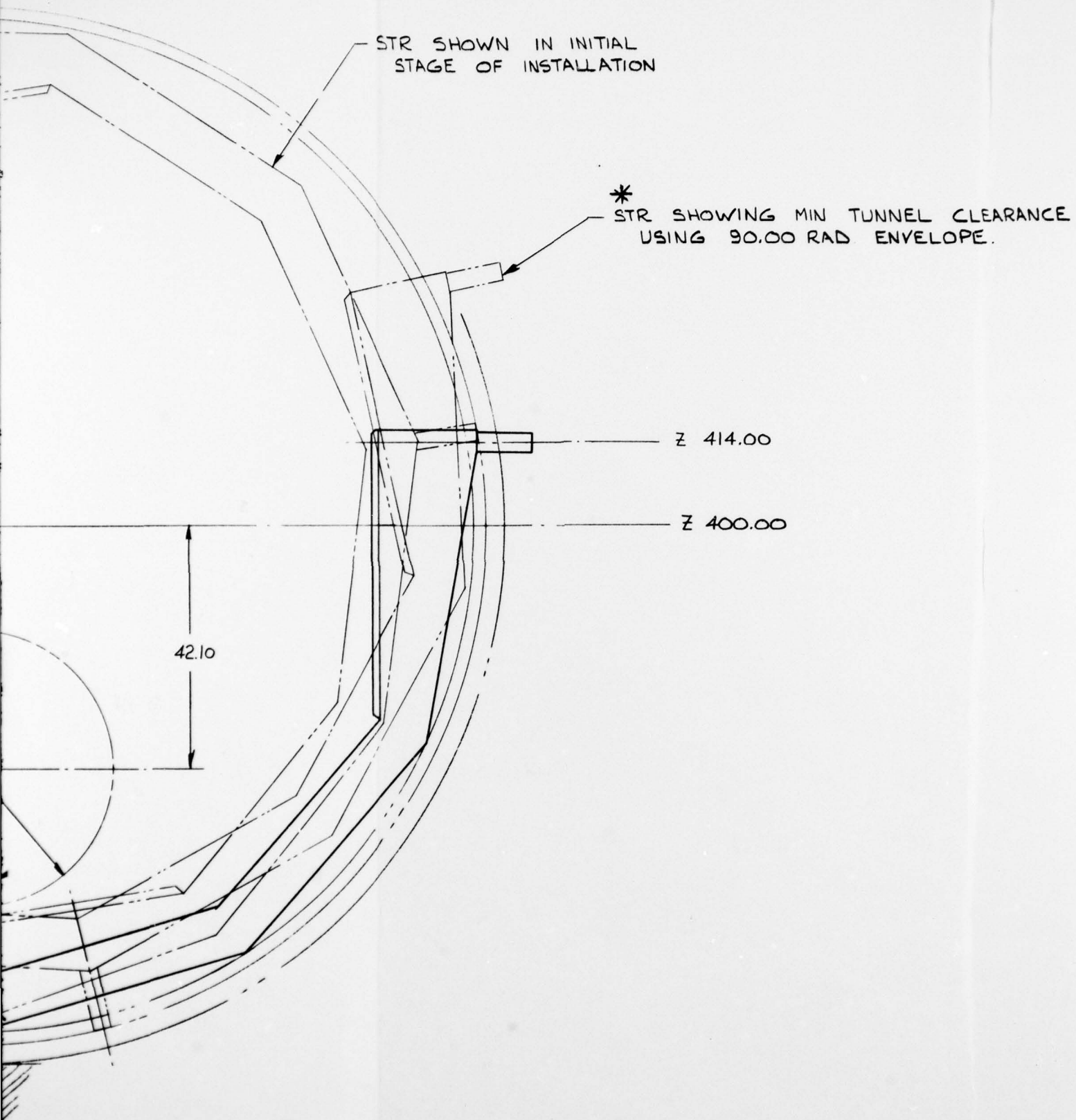


VIEW A-A
SCALE 1/10

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES- TOLERANCES ON 2 PLACE DECIMALS ± / 3 PLACE DECIMALS ± / ANGLES ± / FRACTIONS ±	ALL SURF ✓	SIGNATURES	GENERAL ELECTRIC S S DEPT. VR PHILA, PA.
MATERIAL-			STR./ DEPLOYABLE SPACECRAFT (012) CONFIGURATION
			SIZE / CODE IDENT NO. / STR-011
			SCALE 1/10 / SHEET 1 OF 1

3


87.80 RAD
(STR DESIGN ENVELOPE)
90.00 RAD
(MAX. PAYLOAD ENVELOPE)
TRUNNION BARS (2)
REMOVED THIS SIDE FOR
INSTALLATION CLEARANCE
93.00 RAD
(ORBITER STATIC CLEARANCE)
ENVELOPE
47.40
DIA
TUNNEL
1.50 MIN *
CLEARANCE


STR SHOWN IN INITIAL STAGE OF INSTALLATION
* STR SHOWING MIN TUNNEL CLEARANCE USING 90.00 RAD ENVELOPE.
Z 414.00
Z 400.00
42.10